(2)

NAVAL POSTGRADUATE SCHOOL

Monterey, California

THESIS

AIRCRAFT CONFIGURATION STUDY FOR EXPERIMENTAL 2-PLACE AIRCRAFT AND RPVs

by

Gary Douglas Black

March 1990

Thesis Advisor:	Richard M. Howard
Co-Advisor:	James P. Hauser

Unclassified

Security Classification of this page

REPORT DOCUMENTATION PAGE

1a Report Security Classification Unclassified	1b Restrictive Markings
2a Security Classification Authority	3 Distribution Availability of Report Approved for public release; distribution is unlimited.
2b Declassification/Downgrading Schedule	
4 Performing Organization Report Number(s)	5 Monitoring Organization Report Number(s)
6a Name of Performing Organization Naval Postgraduate School — 6b Office Symbol (If Applicable) 67	7a Name of Monitoring Organization Naval Postgraduate School
6c Address (city, state, and ZIP code) Monterey, CA 93943-5000	7b Address (city, state, and ZIP code) Monterey, CA 93943-5000
8a Name of Funding/Sponsoring Organization — 8b Office Symbol (If Applicable)	9 Procurement Instrument Identification Number
8c Address (city, state, and ZIP code)	10 Source of Funding Numbers

Program Element Number	Project No	Task No	Work Unit Accession No

11 Title (Include Security Classification) AIRCRAFT CONFIGURATION STUDY FOR EXPERIMENTAL 2-PLACE AIRCRAFT AND RPVs

12 Personal Author(s) Black, Gary D.

13a Type of Report	13b Time Covered	14 Date of Report (year, month,day)	15 Page Count
Master's Thesis	From To	1990 March	136

16 Supplementary Notation The views expressed in this thesis are those of the author and do not reflect the official policy or position of the Department of Defense or the U.S. Government.

17 Cosati Codes			18 Subject Terms (continue on reverse if necessary and identify by block number)
Field	Group	Subgroup	Aircraft Design, Remotely Piloted Vehicle, RPV, Configuration, Experimental Aircraft

19 Abstract (continue on reverse if necessary and identify by block number

A performance comparison and tradeoff study was conducted between eight unique aircraft configurations for high performance light aircraft and remotely piloted vehicles. These configurations included conventional tractor, conventional pusher, canard, tandem-wing, joined-wing, and 3-surface designs, which were analyzed though the use of microcomputer-based performance and lattice-vortex programs. Actual experimental aircraft were utilized as models which were scaled to a useful load of 600 pounds and given a common powerplant of 115 horsepower.

The joined-wing, tandem-wing and conventional pusher were found to exhibit sufficient improvement over a conventional tractor configuration to warrant serious consideration for design selection. The performance and stability programs provided reasonably accurate predictions of aircraft performance when given actual aircraft dimensions and power available data and warrant use for preliminary aircraft design.

20 Distribution/Availability of Abstract [X] unclassified/unlimited [ ] same as report [ ] DTIC users	21 Abstract Security Classification Unclassified
22a Name of Responsible Individual Richard Howard	22b Telephone (Include Area code) (408) 646-2870 — 22c Office Symbol Code 67Ho

DD FORM 1473, 84 MAR — 83 APR edition may be used until exhausted — security classification of this page

All other editions are obsolete — Unclassified

Approved for public release; distribution is unlimited.

Aircraft Configuration Study For Experimental 2-Place Aircraft and RPVs

by

Gary Douglas Black
Lieutenant Commander, United States Navy
B.S., Michigan State University, 1976

Submitted in partial fulfillment of the requirements
for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

from the

NAVAL POSTGRADUATE SCHOOL
March 1990

Author: Gary D. Black
Gary Douglas Black

Approved by: Richard M. Howard
Richard M. Howard, Thesis Advisor

James P. Hauser
James P. Hauser, Second Reader

E.R. Wood
E. Roberts Wood
Chairman, Department of Aeronautics and Astronautics

ABSTRACT

A performance comparison and tradeoff study was conducted between eight unique aircraft configurations for high performance light aircraft and remotely piloted vehicles. These configurations included conventional tractor, conventional pusher, canard, tandem-wing, joined-wing, flying-wing and 3-surface designs, which were analyzed through the use of microcomputer-based performance and lattice vortex programs. Actual experimental aircraft were utilized as models which were geometrically scaled to a useful load of 600 pounds and given a common powerplant of 115 horsepower. The joined-wing, tandem-wing and conventional pusher were found to exhibit enough improvement over a conventional tractor configuration to warrant serious consideration for design selection. The performance and stability programs were reasonably accurate predictors of aircraft performance when given actual aircraft parameters and thus judged as reliable estimators of scaled aircraft performance.

TABLE OF CONTENTS

TABLE OF SYMBOLS AND/OR ABBREVIATIONS

A_π	Area upon which an individual component drag area is based
AR	Aspect ratio (main wing only)
c	Specific fuel consumption, lb_f/(ft-lb/sec)sec
c.g.	Center of gravity
C_L	Coefficient of lift
C_D	Coefficient of total drag
C_{Di}	Coefficient of induced drag
CDI2	Induced drag coefficient in POINT performance program
c_d	Coefficient of drag for 2-dimensional airfoil
c_{dmin}	Minimum coefficient of drag for 2-dimensional airfoil
C_{Do}	Coefficient of parasite drag
$C_{D\pi}$	Coefficient of drag based on a revelant area A_π other than wing area
c.g.	Center of gravity
$C_{L\alpha}$	lift-curve slope, $\partial C_L/\partial\alpha$
C_{Ltrim}	Coefficient of lift at trimmed condition
C_L	Coefficient of lift
c_l	Coefficient of lift for 2-dimensional airfoil
$C_{l\beta}$	Rolling moment coefficient due to sideslip, $\partial C_l/\partial\beta$
$C_{M\alpha}$	Pitching moment coefficient due to angle of attack, $\partial C_M/\partial\alpha$
$C_{n\beta}$	Yawing moment coefficient due to sideslip, $\partial C_n/\partial\beta$
DIA	Propeller diameter, ft
BHP	Brake horsepower, manufacturer's specifications

e	Oswald efficiency factor
FAA	Federal Aviation Administration
FG	Fixed gear
H_f	Final altitude
H_i	Initial altitude
H/L	Height/length
hp	Horsepower
k	Induced drag coefficient
KTAS	Knots true airspeed
J	Advance ratio
100LL	100 octane low-lead aviation fuel
lb_f	Pounds of fuel
L_c/-L_t	Canard loading/Tail loading
L/D	Lift-to-drag ratio (C_L/C_D)
mph	Statute miles per hour
n	Propeller speed, rps
NLF	Natural Laminar Flow
nm	Nautical miles
P_A	Power available, ft-lbs/sec
P_R	Power required, ft-lbs/sec
Re	Reynolds number
RG	Retractable gear
ROC	Rate of Climb
rpm	Revolutions per minute
rps	Revolutions per second

RPV	Remotely piloted vehicle
S, S_{ref}	Wing area, ft^2 (includes canard)
S_c	Canard area
S_t	Tail area
SFC	Specific fuel consumption (lbf/BHP/hr)
V	True airspeed , ft/sec
V_{max}	Maximum true airspeed , ft/sec or mph
V_x	Velocity of maximum angle of climb
V_y	Velocity of maximum rate of climb
X_{ac}	Neutral point
X_{ref}	Designated datum point in LINAIR
W_0	Weight at beginning of leg
W_1	Weight at end of leg
W/L	Wing loading or width/length
W_L	Landing weight
W_{TO}	Takeoff weight
α	Angle of attack
β	Sideslip angle
δ	Correction factor for deviation from elliptical lift distribution
λ	Scaling factor
η	Propeller efficiency
ρ_∞	Freestream density of air
τ	Correction factor for deviation from elliptical lift distribution

ACKNOWLEDGMENTS

When I first arrived at the Naval Postgraduate School with virtually no math or engineering background and saw the Aeronautical Engineering syllabus, I set my goal to last as long in the curriculum as possible. Fortunately, there were numerous others intent on seeing me all the way through. Professor Hamming, who could have taught any doctoral-level computer course anywhere, instead taught my calculus refresher course in the manner of Professor Kingsfield from *The Paper Chase* and laid the foundation for a solid math background. Professor Lal, who first recognized how much I needed to learn to catch up in my first quarter and spent numerous hours in personal tutoring to help me understand engineering, eventually graded my first perfect exam and made the end result seem attainable. I sincerely appreciate the patience of each and every professor in the Aeronautical Engineering Department for their endless patience in explaining solutions and filling in the "missing steps." I salute your profession and dedication. In particular, I appreciate the guidance and enthusiasm of Professor Rick Howard, who always had time to provide assistance, be it at 11 P.M. or on a Sunday morning. I also thank Martin Hollmann for his suggestions and assistance in my thesis study. This degree definitely would not have been possible without the teamwork, competition and friendship of my classmates. Fly safe and I wish you well.

Lastly, I thank our parents for their constant support and encouragement. Most of all, I thank my wife Celeste for the endless times she sent me down to study and for the extra hours she dedicated at home in addition to her job. We did it!

I. INTRODUCTION

In 1903, Orville Wright was credited with the first powered heavier-than-air aircraft flight in a canard-configured airplane. However by 1920, the conventional configuration of a main forward wing with an aft horizontal tail surface had become dominant in aircraft design. The predominance of the conventional configuration may be attributed to two historical factors. The "dogfighting" requirements of World War I dictated that fighter aircraft be highly maneuverable. Tailplanes provided agility whereas canards were overly stable and prevented stalls and spins. Secondly, the heavy weight and size of early aircraft engines severely limited the center of gravity locations for which a canard-configured pusher aircraft could compensate [Ref. 1:p. 8]. The answer in those days was to mount the engine in front and balance the aircraft with an aft horizontal tail. Although some hybrid designs existed, none was successful enough to be considered a breakthrough. World War II and the post-war years provided some unique designs in the form of flying wings by Messerschmitt (Me-163) and Northrop (YB-49), forward swept-wings by Junkers (Ju-287) and multi-fuselaged aircraft such as the North American Twin Mustang (P-82), but little else other than conventional aircraft for commercial and general aviation use.

Decades passed without a highly successful non-conventional design until Burt Rutan introduced a small two-place fiberglass aircraft with a canard-configuration called the "VariEze" in 1975. The VariEze, due to its low cost, simplicity of construction and high speed on low horsepower, resulted in hundreds of orders for plans by experimental aircraft homebuilders and many

similar follow-on designs. With the advent and widespread use of composite materials for aircraft construction, a large variety of aircraft configurations appeared in kitplane and aviation trade magazines. Aircraft like the Beech Starship, OMAC 1 and Avtek 400 touted the advantages of canards for commercial application, as did the Gates-Piaggio Avanti for 3-surface aircraft [Ref. 1:p. 8]. Beechcraft claimed enhanced stability and stall resistance with its canard versus conventional configuration as well as increased fuel efficiency, improved lift-to-drag ratios and increased wing efficiency and effective span with its tipsails or winglet rudders [Ref. 2:p. 9]. Piaggio claimed its Avanti would fly 60 mph faster, take off 3490 pounds lighter and carry the same number of passengers on 30 percent less horsepower than the Starship [Ref. 3:p 37]. As will be seen later, both the canard and 3-surface configurations compared very favorably against conventional twin-engine business aircraft. Some other non-conventional aircraft, such as the Learfan, a twin-engine pusher single-propfan configuration with a conventional wing layout, failed more as a result of financing during FAA certification than as a result of design [Ref. 4:p. 82].

The proliferation of exotic designs for the experimental aircraft builder and for application to military remotely piloted vehicles (RPVs) could be attributed to the ability to utilize layup molds and composite materials. Fiberglass cloth is simple and inexpensive to work with and conducive to blended-wing bodies and complex curves. Graphite fibers have provided greater stiffness and lighter weight in structures previously constructed of aluminum and wood without the complex joints and fasteners required of them.

With such a panorama of design choices, how can a fledgling aircraft designer go from "back of the napkin" sketches to full-scale prototypes without the availability of wind tunnels, supercomputers and significant financial backing? A small sample of possible configurations is shown in Figures 1 and 2.

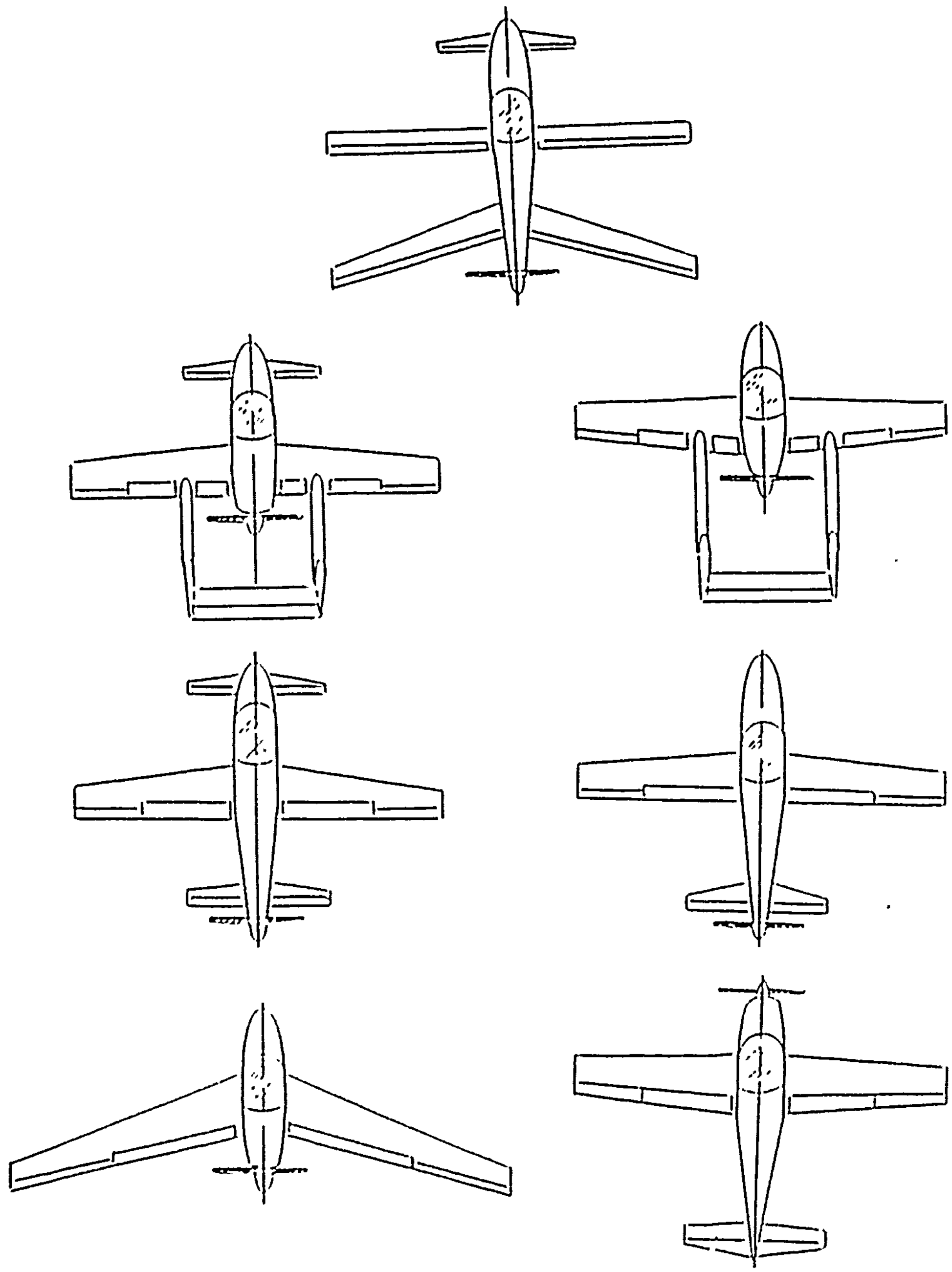


Figure 2. 3-Surface, Flying Wing and Conventional Configurations

Likewise, how can an aircraft user equitably compare the presently available designs with passenger loads varying from 1 to 4 or more, engines ranging from 26 hp to 300+ hp, and wing areas and loadings also spanning from low to high? One solution lies in the use of simple microcomputer-based programs for predicting aircraft performance and handling qualities and of the concept of scaling aircraft geometrically.

This thesis examined nine different aircraft configurations and made a comparative study of the performance advantages and disadvantages of each. The various configurations and the aircraft designs used for reference were:

• Conventional tractor retractable	Lancair 235
• Conventional tractor fixed gear	RV-4
• Conventional pusher	Mini-Imp
• Canard	LongEze
• Tandem-wing (Low W/L)	Dragonfly
• Tandem-wing (High W/L)	Q-200
• Joined-wing	Ligeti Stratos
• Three Surface	Discovery
• Flying Wing	Mitchell U-2

Descriptions of the full-scale aircraft may be found in the Appendix by aircraft type. The reader may wish to briefly familiarize him or herself with the aircraft design as the aircraft are referred to by configuration in the remainder of the thesis.

Each of the configurations listed above has been successfully flown, but head to head comparison was "fruitless" since "apples were not being compared to apples." A trade study comparison of the full scale aircraft was not legitimate since the various aircraft used different powerplants, had unequal useful loads,

and incorporated unique design constraints. These aircraft were similar in one regard, in that they were designed to carry one or two people at a high speed (150+ mph) on a low horsepower engine (160 hp or less).

This study made the comparisons between the configurations by standardizing the powerplant and scaling the aircraft listed above to carry the same useful load. Although each of the full scale aircraft had been successful in flight test, these scaled aircraft are untested except by computer prediction using theoretical equations. These equations are the same equations studied in introductory Aeronautical Engineering courses such as AE 2035 (Basic Aerodynamics) and AE 2036 (Performance, Stability and Control). The computer methods of determining the drag polar and thus the remainder of flight performance characteristics are similar to the methods studied in AE 4323 (Flight Evaluation Techniques) and AE 4273 (Aircraft Design).

The conclusions of this study may be utilized as a starting point for an aircraft designer with specific aircraft traits or requirements in mind. Once the possibilities are narrowed down to a single configuration then the designer can begin the task of optimization and compromise.

This report was prepared as a partial requirement for an Aeronautical Engineering degree. The author, academic advisor and the U. S. Navy assume no legal liability or responsibility for the accuracy of the performance predictions or assumptions contained within this report. The aircraft utilized in this study are based on information packets available to the general public and performance reports published in aviation trade magazines. Because of the unique designs of the representative aircraft, tradenames have been used for reference and do not constitute an endorsement of any product or designer.

II. BACKGROUND

A. AERODYNAMIC TRADEOFF STUDIES OF CONVENTIONAL, CANARD, AND 3-SURFACE AIRCRAFT

In 1984, Kendall and Gates Learjet Corporation conducted a study using the theorems of Prandtl and Munk to show that the 'ideal' minimum induced drag could be achieved with a modern three-surface airplane in trim if equal and opposite vertical loads were applied by the forward and aft trimming surfaces [Ref. 5]. Kendall compared these results with the theoretical results for conventional and canard-configured aircraft. For two-surface aircraft, the minimum induced drag required a zero trim surface load. This could only be attained at one aft c.g. location on the conventional aircraft and was not attainable on the canard configuration without losing positive static longitudinal stability. Therefore, two-surface aircraft must be designed for minimum induced drag at some average c.g. location. Since canard uploads were generally higher than tail downloads, the conventional configuration typically had lower induced drags. Conversely, 3-surface aircraft using both fore and aft trimming surfaces could trim for minimum induced drag at any c.g. location. Thus, Kendall summarized that the 3-surface design could have better cruise efficiency in a stable trimmed condition over a practical range of c.g. locations, followed by the conventional design and lastly, the canard configuration [Ref. 5:p. 9].

In 1985, Selberg and Rokhsaz determined the theoretical induced and viscous drag under trimmed conditions between conventional, canard and 3-surface aircraft configurations [Ref. 6]. A three surface vortex lattice method was used

to trim the aircraft as well to predict the induced drag of each configuration. To predict the inviscid and viscous characteristics, a vortex panel method was used in conjunction with a momentum integral boundary layer method. The aircraft were modeled after modern day 6-passenger business turboprops. The useful load was constant at 1200 pounds with a maximum gross weight of approximately 4600 pounds. The main wing area was 120 square feet with an aspect ratio of 12. All configurations utilized the same vertical tail and the same NASA MS(1)-0313 airfoil for all lifting surfaces. The top views of the three configurations are shown in Figures 3-5.

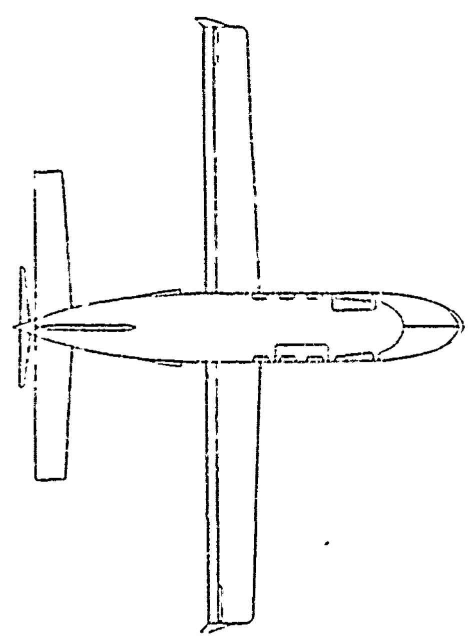


Figure 3. Conventional 6-place Business Turboprop [Ref. 6]

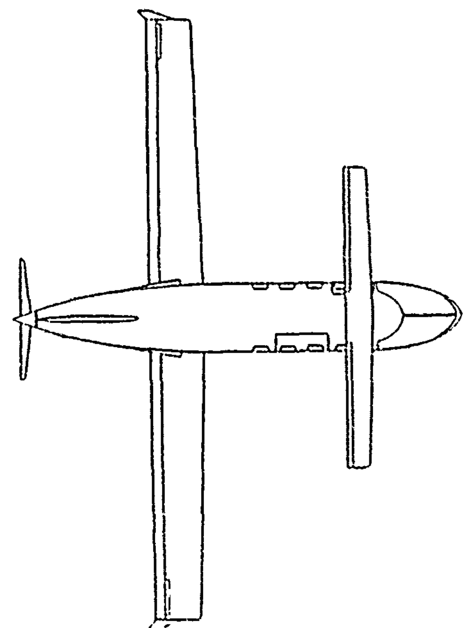


Figure 4. Canard 6-place Business Turboprop [Ref. 6]

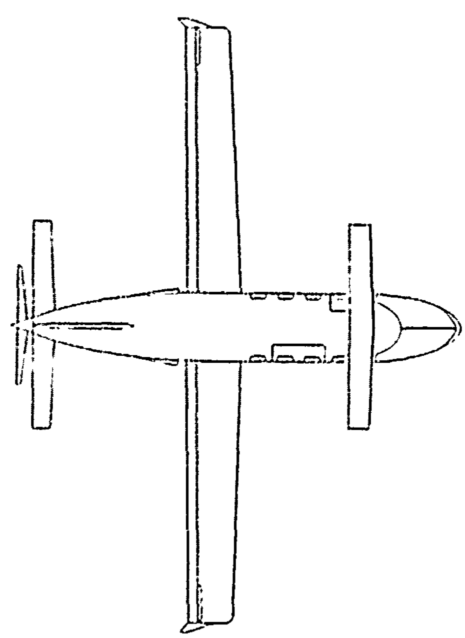


Figure 5. Tri-surface 6-place Business Turboprop [Ref. 6]

The parameters varied included static marg ·, stabilator surface area, stabilator loading ratios, and aspect ratio. Aspect ratios were limited such that they would be feasible for all-composite lifting surfaces. For the 3-surface design, the canard and tail had equal areas. The objective of the paper was to analyze the behavior of the ratio C_{Ltrim}/C_{Di} over a range of static margins and area ratios. C_{Ltrim}/C_{Di} is the ratio of total lift coefficient at trim to the induced drag coefficient and is the primary indicator of flight efficiency for range, endurance and fuel economy and therefore of great interest to business-class aircraft designers.

The study found that for the 3-surface configuration, the maximum value of C_{Ltrim}/C_{Di} occurred at approximately a ratio of the canard load to negative tail load of 2.0. C_{Ltrim}/C_{Di} degraded rapidly for canard to negative tail load ratios less than 2.0, but the effect was not as severe for higher values as shown in Figure 6.

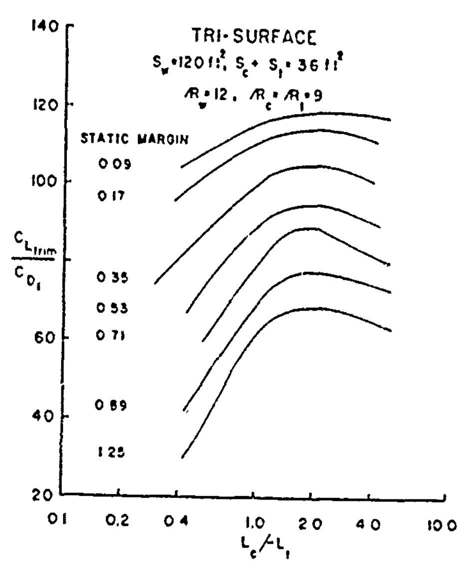


Figure 6. Tri-surface Induced Drag Sensitivity to the Ratio of Canard to Tail Loading and Static Margin [Ref. 6]

Both the conventional and canard configurations were evaluated as a function of tail or canard area as shown in Figures 7 and 8.

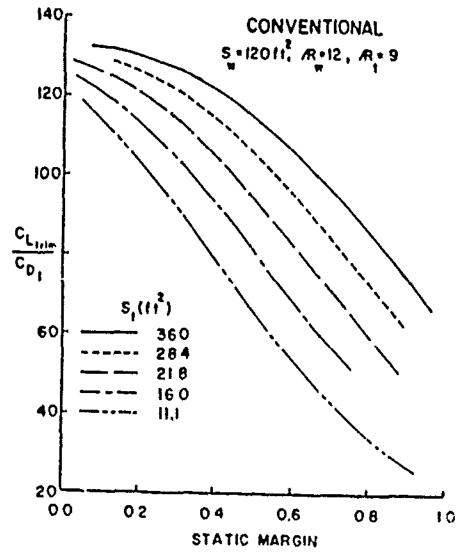


Figure 7. Conventional Configuration Induced Drag Sensitivity with Changes in Tail Area and Static Margin [Ref. 6]

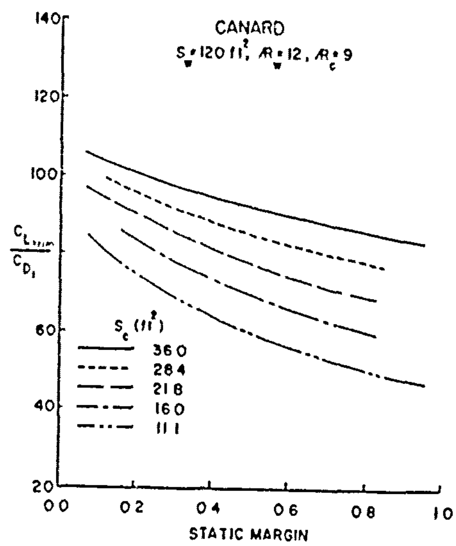


Figure 8. Canard Configuration Induced Drag Sensitivity with Changes in Canard Area and Static Margin [Ref. 6]

A comparison of the graphs at a common static margin of 0.35 shows that the conventional configuration had the highest C_{Ltrim}/C_{Di} with a range between 85 and 125, whereas the canard had the lowest C_{Ltrim}/C_{Di} ratio ranging from 66 to 96. Enlarging the tail size increased C_{Ltrim}/C_{Di} because of the lower C_L of the tail and hence the lower induced drag. The 3-surface configuration C_{Ltrim}/C_{Di} ratio varied between 100 and 104 for canard-to-tail loadings of 1.0 to 4.0. The study found that the 3-surface aircraft was inferior to conventional aircraft in terms of induced drag for static margins less than 0.85. However, the 3-surface configuration was limited to equal areas for the canard and tail surface area and this may not have been the optimum ratio. The canard configuration had the highest C_{Ltrim}/C_{Di} for higher static margins, but this was not a factor since most business class aircraft have a static margin between 0.25 and 0.43 [Ref. 6:p. 6]. For peak span efficiencies, a canard aspect ratio to wing aspect ratio of 1.5 to 2.0 was required. One should note that induced drag reduction is not the only measure of improved flight efficiency. The goal of decreased total drag reduction may not occur with decreased induced drag reduction, as parasite drag will increase as the wing and tail areas are enlarged.

Additional tests were conducted using the natural laminar flow airfoil, NASA NLF-0215F, and the NACA 23012 airfoil. The NLF airfoil raised the overall C_{Ltrim}/C_{Di} results whereas the NACA 23012 airfoil lowered the results. In all cases, at normal static margins and lower stabilator aspect ratios, the conventional configuration had the highest lift-to-drag ratio with those for the canard and 3-surface about equal. The highest stabilator surface areas produced the least induced drag for all configurations. However, Selberg and Rokhsaz concluded the overall C_{Ltrim}/C_{Di} was close enough between all the configurations

that configuration selection should probably be made from other considerations, such as stability and control, safety, structural requirements and costs [Ref. 6:p. 9].

The Selberg and Rokhsaz study compared the ratios of C_{Ltrim}/C_{Di} for the three configurations and predicted the conventional configuration to be slightly superior to the 3-surface configuration which was judged to be superior to the canard configuration. The Kendall study used theory to predict the minimum induced drag would be attained with the 3-surface aircraft, followed by the conventional aircraft and lastly the canard configuration. These studies were similar, but differed in the parameters being optimized. The value of the minimum induced drag may be not correspond to that for the minimum value of C_{Ltrim}/C_{Di}. Therefore, the studies were not comparative for configuration selection.

B. COMPARATIVE STUDIES OF CONVENTIONAL, CANARD, AND 3-SURFACE AIRCRAFT

A comparison conducted by a major aviation trade magazine between the Piaggio Avanti, a 3-surface configuration shown in Figure 9, the Beech Starship, a canard-configured twin turboprop shown in Figure 10, and several conventionally-configured twin turboprop business-class aircraft tends to justify the results of the two studies [Ref 7].

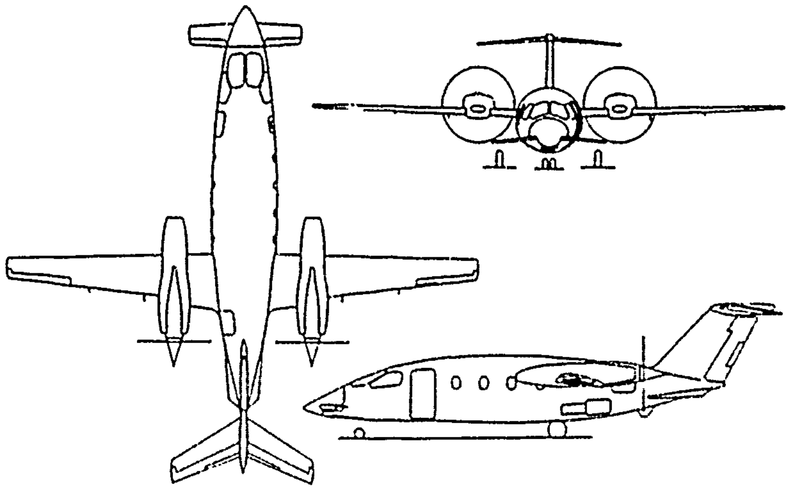


Figure 9. Piaggio Avanti [Ref. 8:p. 86]

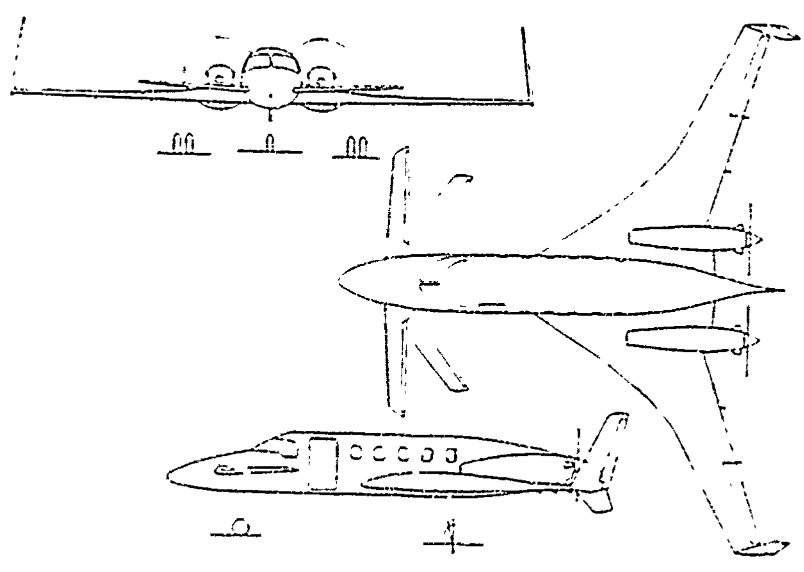


Figure 10. Beech Starship [Ref. 8:p. 81]

Although the Starship has a maximum takeoff weight of 14,400 pounds and the Avanti maximum takeoff weight is 10,810 pounds, both aircraft have similar payload-range characteristics and their cabin volumes are both roughly 500 cubic feet In spite of the Avanti having a higher cruise airspeed of 391 KTAS versus the Starship's 315 KTAS, the Avanti had a better specific range of 0.82 nm/lb of fuel to the Starship's 0.55 nm/lb. In this comparison, the 3-surface outperformed the two conventional twin turboprop aircraft in its class as well, as shown in Figure 11.

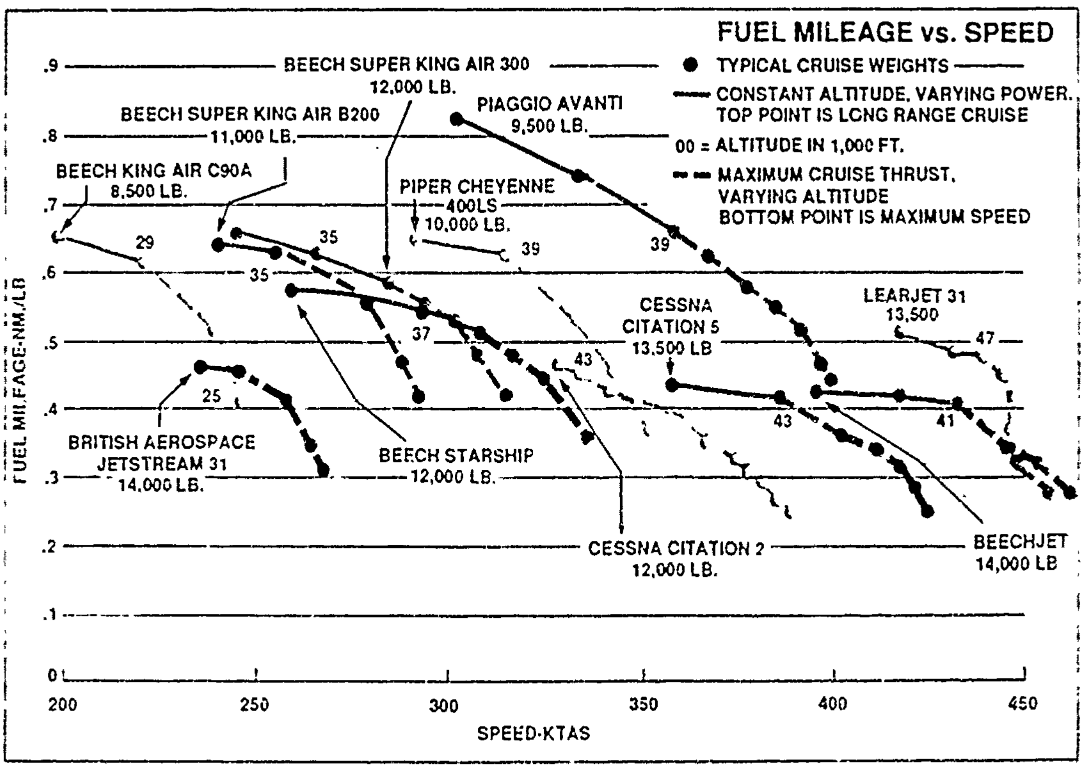


Figure 11. Twin Turboprop Comparison Chart [Ref. 7:p. 75]

This chart demonstrates that comparison of actual aircraft and prototypes may be of as much usefulness to aircraft designers as pure theoretical studies. In contrast to the Selberg and Rokhsaz study, the canard aircraft is quite equitable with conventional aircraft in its class whereas the 3-surface Piaggio Avanti significantly outperformed its competitors in fuel economy and speed for turboprops. Therefore, this thesis utilized actual aircraft or prototypes that have successfully flown for comparison in the 2-place aircraft category.

III. PROCEDURE

The number of variables affecting aircraft performance is enormous; therefore a few parameters were held constant and others were limited. Since the ultimate design specification of any aircraft is its useful load, this was held constant at 600 pounds. This equated to two aircrew at 170 pounds each, approximately 32 gallons of fuel and 50-60 pounds of baggage. This same useful load would provide approximately 400 pounds of monitoring and remote control equipment for an RPV. For four passengers or up to 1000 pounds of RPV equipment, the same configuration performance values could be scaled appropriately. The total wing areas for each aircraft ranged between 67-216 square feet and the gross weight varied between 1025 and 1575 pounds. Wing area was defined as the total area of lifting surfaces, which included the canards on the canard, 3-surface, tandem and joined-wing configurations.

Each aircraft configuration utilized the identical aircraft engine, and the propeller efficiency was also common. A power available versus true velocity curve at constant rpm was plotted for this engine and propeller combination and used in the computer performance program POINT written by Dr. Frederick O. Smetana [Ref. 9]. Users may select any engine and propeller combination by following this procedure.

The following performance parameters were computed for each scaled aircraft and then compared with one another:

- Gross weight
- Flat plate drag area
- Parasite drag coefficient

- Maximum climb angle and speed
- Maximum climb rate and speed
- Maximum level flight speed
- Speeds for max endurance and max range
- Service and absolute ceilings
- Max range and endurance for a given fuel load
- Stability derivatives ($C_{L\alpha}$, $C_{m\alpha}$, $C_{l\beta}$ and $C_{n\beta}$)

The lift, moment, rolling and yawing coefficients were computed by the lattice vortex program LINAIR by Dr. Ilan Kroo [Ref. 10]. This program calculates the aerodynamic characteristics of multi-element nonplanar lifting surfaces by solving the Prandtl-Glauert equation for inviscid, irrotational, subsonic flow. Once LINAIR has solved the system of equations, it utilizes the Kutta-Joukowski relation to compute the forces and moments acting on the configuration. LINAIR is a simplified version of the lattice vortex program demonstrated in AE 3501 (Advanced Aerodynamics).

The 3-view drawings of the aircraft on which the configurations are based and the output from the POINT performance and LINAIR programs may be found in the Appendix by aircraft type.

IV. POWER AVAILABLE

In order to equitably compare the various aircraft configurations, the power available to each aircraft must be equivalent. The powerplant chosen for this study was the Avco Lycoming Model O-235-L2A. This engine is very common in the general aviation fleet, readily accessible to experimental homebuilders and relatively inexpensive for use on RPV's or targets. The O-235-L2A is a direct-drive, four-cylinder, horizontally-opposed, air-cooled engine. This engine is supplied at the factory with an automotive type alternator and starter. The O-235-L2A has a rated maximum continuous power of 118 hp at 2800 rpm at standard sea level conditions using grades 100/100LL aviation fuel. This engine has an alternate rating of 115 BHP at 2700 rpm which was used for this study [Ref. 11:p. 1]. This alternate rating was used since most aircraft owners tend to use less than full power for enroute climb and cruise for increased engine life and economy.

The power available curve which was used for all the aircraft configurations was constructed by multiplying the brake horsepower by a propeller efficiency as a function of velocity. A generic 62-inch diameter propeller was utilized with performance data obtained for the two-blade Clark-Y section propeller 5868-9 which was tested and documented in *NACA Report 640* [Ref. 12]. It should be noted that many combinations of propeller diameter, pitch and shape could be tested for each aircraft until the optimal performance is obtained. Therefore, holding the propeller diameter and pitch constant in this comparison was justified. The advance ratio (J) was tabulated as a function of velocity by holding

the rpm and diameter constant at 45 rps and 62 inches respectively as shown in the following equation:

$$J = \frac{V(\frac{ft}{sec})}{n(\frac{rev}{sec}) \times DIA\ (ft)} \tag{1}$$

Normally the engine rpm would increase as the aircraft increases velocity as in a slight dive, but for this efficiency calculation, the study assumed the pilot incrementally throttled back to hold rpm constant. The design chart for the Clark-Y section, two-bladed propeller 5868-9 is shown in Figure 12. This information may also be available from the propeller manufacturer for a specific fixed-pitch or constant-speed propeller.

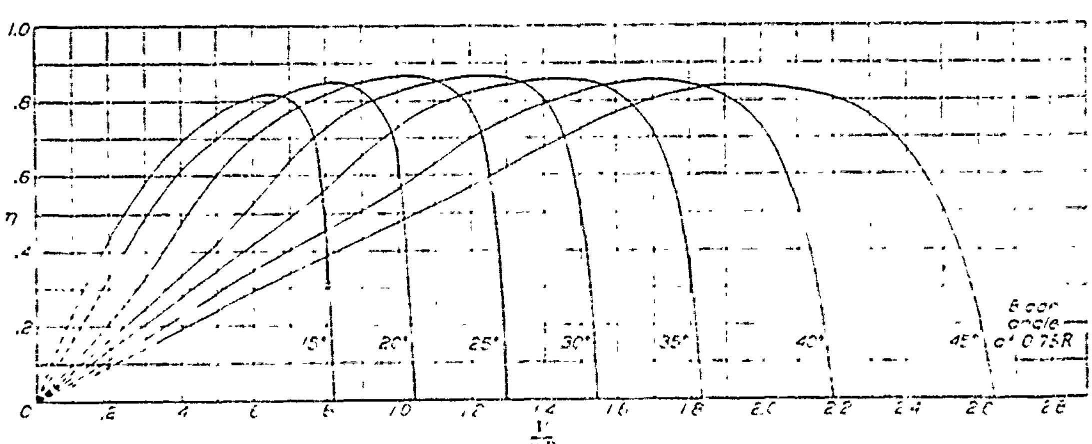


Figure 12. Design Chart for Propeller 5868-9, Clark-Y section, 2 blades [Ref. 12:p. 550]

In order to obtain η for the 62-inch propeller, seven points were selected off the 30° blade pitch curve. The 30° blade pitch was chosen since it gave the highest propeller efficiency at airspeeds in the 180-190 mph range where the

design aircraft are expected to cruise. These seven points were replotted and a 3rd order polynomial curve was fitted to give an equation for η as a function of J as shown in Figure 13.

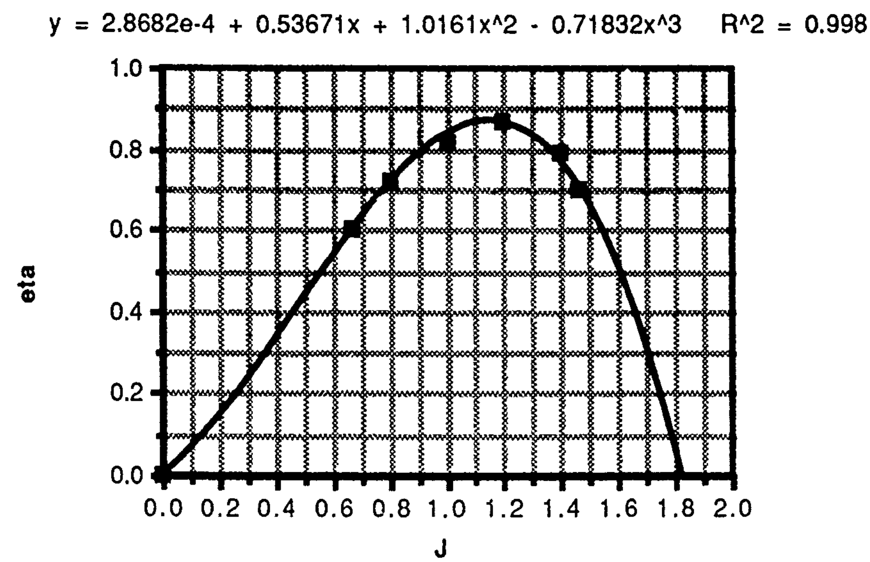


Figure 13. Propeller Efficiency versus Advance Ratio

Now, with values of η as a function of J and therefore of velocity, the actual power available (not counting accessories) could be calculated through the use of a spreadsheet.

Therefore, the actual power available with this propeller was:

$$\text{Power}_{\text{avail}}\left(\frac{\text{ft.lbs}}{\text{sec}}\right) = \text{BHP} \times 550\frac{\text{ft.lbs}}{\text{hp}} \times \eta \tag{2}$$

These values were computed in Table 1 and were used to generate the power available versus velocity curve shown in Figure 14.

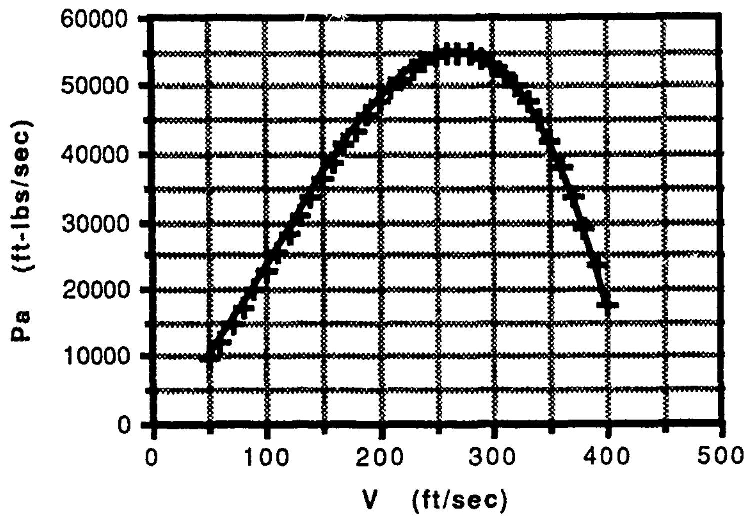


Figure 14. Power Available versus Velocity

Figure 15 shows the same power available in the common units of horsepower and miles per hour.

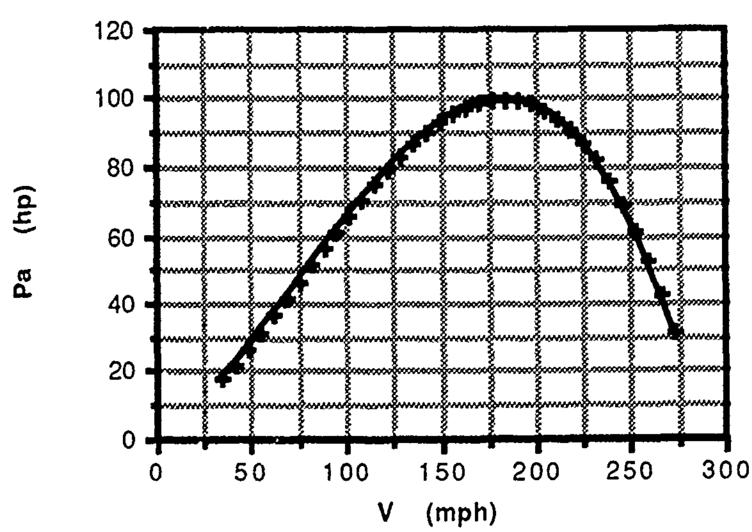


Figure 15. Power Available versus Velocity

The bold values in Table 1 were used in the POINT program to provide a power available curve for computer-generated aircraft performance prediction. Intermediate values of η were also calculated for velocities between 60 and 300 ft/sec in steps of 2 ft/sec for use in estimating maximum range and endurance. These power available and propeller efficiency data were standard for all of the configurations, although they slightly favored the aircraft designed with cruise speeds of 180-190 mph. The aircraft designer using the format of this thesis should select an advance ratio with peak efficiencies near the desired cruise speed of his or her design.

TABLE 1. POWER AVAILABLE DATA FOR THE O-235 ENGINE

Prop Dia (ft) = 5.17 BHP = 115 rho/rho 0 = 1 N (rpm) = 2700

V (ft/sec)	V (mph)	J	η	Pa (ft-lbs/sec)	Pa (hp)
50.0	34.0	.215	.155	9831	17.9
60.0	40.8	.258	.194	12268	22.3
70.0	47.6	.301	.234	14813	26.9
80.0	54.4	.344	.276	17443	31.7
90.0	61.2	.387	.318	20138	36.6
100.0	68.0	.430	.362	**22875**	41.6
110.0	74.8	.473	.405	25634	46.6
120.0	81.6	.516	.449	28391	51.6
130.0	88.4	.559	.492	31127	56.6
140.0	95.2	.602	.535	33818	61.5
150.0	102.0	.645	.576	36444	66.3
160.0	108.8	.688	.616	38983	70.9
170.0	115.6	.731	.655	41413	75.3
180.0	122.4	.774	.691	43712	79.5
190.0	129.3	.817	.725	**45859**	83.4
200.0	136.1	.860	.756	47832	87.0
210.0	142.9	.903	.784	49610	90.2
220.0	149.7	.946	.809	51170	93.0
230.0	156.5	.989	.830	52492	95.4
240.0	163.3	1.032	.847	53553	97.4
250.0	170.1	1.075	.859	54333	98.8
260.0	176.9	1.118	.867	54808	99.7
270.0	183.7	1.161	.869	54958	99.9
280.0	190.5	1.204	.866	**54762**	99.6
290.0	197.3	1.247	.857	54196	98.5
300.0	204.1	1.289	.842	53240	96.8
310.0	210.9	1.332	.820	51872	94.3
320.0	217.7	1.375	.792	50071	91.0
330.0	224.5	1.418	.756	47814	86.9
340.0	231.3	1.461	.713	45081	82.0
350.0	238.1	1.504	.662	41849	76.1
360.0	244.9	1.547	.602	38096	69.3
370.0	251.7	1.590	.534	33802	61.5
380.0	258.5	1.633	.458	**28944**	52.6
390.0	265.3	1.676	.372	23501	42.7
400.0	272.1	1.719	.276	17452	31.7

V. SCALING FACTORS

For each aircraft configuration, the useful load was held constant at a value of 600 pounds. This equated to two aircrew at 170 pounds each, 32 gallons of fuel at six pounds per gallon, and 68 pounds of baggage. An RPV with the same fuel load could carry over 400 pounds of monitoring and remote control equipment and keep it airborne for approximately 5 hours.

The method of scaling each aircraft design was adapted from a 1950 study on the use of models for flight testing and conversion of data to the full-scale aircraft. The flight test engineer derived the scaling principles based on the theory that the model reacts to all the same forces, both known and unknown, in the proper magnitude, direction and sequence as the full-scale aircraft [Ref. 13:p 457]. The scaling factors used in this study were published in an article for building scale model aircraft that could be used for flight test [Ref. 14:p. 30]. The data collected could only be considered valid if the models were **dynamically similar** to the actual aircraft to be flown. This includes the propulsive power, the weight , and the distribution of the weight. Although a 1/5 scale model might have a wing span that is 1/5 as long as the full scale aircraft, the scale weight is not 1/5 the full scale weight, but rather 1/125 since the weight varies as the cube of the linear dimensions. Thus **dimensional conversions** were required and symbolized by the use of λ,the scaling factor, as shown in Table 2.

TABLE 2. SCALING FACTORS [Ref. 14:p. 34]

$$\lambda = \frac{\text{Full Scale Linear Dimensions}}{\text{Model Linear Dimensions}}$$

Model Design

Parameter	**Model Should Be:**	**Example for 1/5 Scale Model ($\lambda = 5$) Full Scale**	**Model**
Linear Dimensions	Full Scale/λ	Span: 35.8 Ft.	35.8/5 = 7.16 Ft.
Area	Full Scale/λ^2	Wing: 174 sq. ft.	174/25 = 6.96 sq. ft.
Volume, Mass, Force	Full Scale/λ^3	Gross Wt. = 2645 lbs.	2645/125 = 21.16 lbs.
Moment	Full Scale/λ^4		Full Scale/625
Moment of Inertia	Full Scale/λ^5	Pitch: 1346 slug ft.^{2}	1346/3125 = 0.431 slug ft.^{2}
Linear Velocity	Full Scale/$\sqrt{\lambda}$	Max: 144 mph	144/2.24 = 64 mph
Linear Acceleration	Same as Full		Same as Full Scale
Angular Acceleration	Full Scale x λ		Full Scale x 5
Angular Velocity	Full Scale x $\sqrt{\lambda}$		Full Scale x 2.24
Time	Full Scale/$\sqrt{\lambda}$		Full Scale/2.24
Work	Full Scale/λ^4		Full Scale/625
Power	Full Scale/$\lambda^{3.5}$	Rated: 160 hp	160/280 = 0.57 hp
Wing Loading	Full Scale/λ	15.2 psf	15.2/5 = 3.04 psf
Power Loading	Full Scale x $\sqrt{\lambda}$	16.5 lbs./hp	16.5 x 2.24 = 37 lbs./hp
Angles	Same as Full		Same as Full Scale
R.p.m.	Full Scale x $\sqrt{\lambda}$	Rated: 2750 rpm	2750 x 2.24 = 6160 rpm

Full Scale Performance from Model Test

Parameter	**Full Scale Should Be:**	**Example for 1/5 Scale Model ($\lambda = 5$) Measured Model Perf.**	**Derived Full Scale Perf.**
Time	Model x $\sqrt{\lambda}$		Model x 2.24
Maximum Speed	Model x $\sqrt{\lambda}$	64 mph	64 x 2.24 = 144 mph
Max. Climb Rate	Model x $\sqrt{\lambda}$	344 fpm	344 x 2.24 = 770 fpm
Takeoff Distance	Model x λ	160 ft.	160 x 5 = 800 ft.
Pitch, Roll & Yaw Rates	Model $\sqrt{\lambda}$	50°/sec.	50/2.24 = 22°/sec.

For this aircraft configuration study, λ was not a given value, but was determined for each aircraft. This study scaled the aircraft by useful load or weight. Since weight goes as volume, it varies as the cube of λ. The actual aircraft useful load was known as well as the desired useful load of 600 pounds. Therefore λ was backed out as follows:

Example actual aircraft useful load: 400 lbs

$$\lambda^3 = \frac{\text{Actual useful load}}{\text{Desired useful load}} = \frac{400}{600} = \frac{2}{3} \tag{3}$$

$$\lambda = \sqrt[3]{\frac{2}{3}} = 0.8736 \tag{4}$$

A spreadsheet was set up to properly scale all required parameters by taking the cube root of the useful load ratio and using this λ value as shown in Table 3. Each aircraft had a different value of λ which was used to scale the remainder of the dimensions and parameters. These scaled dimensions were used in the performance programs as required. The performance values were scaled only for the sake of interest since these values are based on the aircraft's installed engine and not the common powerplant.

The major concern with scaling aircraft which cannot be accounted for easily is the effect of the Reynolds number In this study, the difference between the Reynolds number of the actual and of the scaled aircraft was less than 5% for most of the aircraft. In the study's worst case where λ equaled 0.74, the Reynolds numbers were still quite similar and for Re between 2 and 4 million, there is no significant difference in the effects. For example, the Re for the scaled joined-wing aircraft based on the chord length are:

$$Re_{actual} = \frac{0.00238 \times 176.4 \text{ ft/sec} \times 2.54 \text{ ft}}{3.751E-7} = 2.843E6 \tag{5}$$

$$Re_{mod\ cl} = \frac{0.00238 \times 176.4 \text{ ft/sec} \times 3.44 \text{ ft}}{3.751E-7} = 3.850E6 \tag{6}$$

The lift and drag values for a Re of 2.84 million is not significantly different from those at a Re of 3.85 million.

TABLE 3. SCALED AIRCRAFT PARAMETERS BASED ON λ

Conventional retractable tractor aircraft based on the Lancair 235

Aircraft Specifications		λ = **0.99**
	Actual values	Scaled values
Wing span (ft.)	23.40	23.67
Wing chord root	4.33	4.38
Wing chord tip	2.38	2.40
Wing area (sq. ft.)	76.00	77.74
Wing airfoil section	NLF-0215-F	NLF-0215-F
Wing aspect ratio	7.20	7.20
Wing loading (GW) (lb./sq. ft.)	18.42	18.63
Effective horizonal tail span (ft.)	6.67	6.74
Horizonal tail chord root	2.17	2.19
Horizonal tail chord tip	1.58	1.60
Horizonal tail area (sq. ft.)	10.60	10.84
Horizonal tail airfoil section		
Horizonal tail aspect ratio	4.19	4.19
Vertical tail area	8.60	8.80
X-section height incl canopy (ft)	3.25	3.29
X-section height (firewall)	2.50	2.53
X-section width	3.67	3.71
Length overall (ft.)	20.00	20.23
Fuel capacity (usable)	33.00	
Empty weight (lb.)	820.00	848.28
Gross weight	1400.00	1448.28
Useful load	**580.00**	**600.00**
Wheel base (in.)	48.00	48.55
Nose wheel	retractable	
Advertised performance		
Max speed sea level (mph)	225	223.73
75% cruise speed 8k'	210	208.82
Stall speed	55	54.69
ROC sl (ft/min) (GW)	1150	1143.52
Range at 75% (no reserve) nm	1000	
G limits at gross weight	9/-4.5	
Horsepower	118	122.76

VI. DRAG POLAR

The POINT performance program required an estimated drag polar as an input in the form of $C_D = C_{D_O} + k\,C_L^2$. This study utilized the $C_{D\pi}$ method as presented in *Aircraft Performance, Stability and Control* [Ref.9:p. 40] to estimate the parasite drag coefficient C_{D_O}. The $C_{D\pi}$ method cannot account for the increased profile drag experienced by the aircraft at large angles of attack and therefore was not used to predict stall speeds.

In this method, the drag coefficient of each component is based on an area, A_π, which is appropriate for that component. Airfoil-surface areas are based on their surface area whereas all other components are based on frontal cross-section areas. For example, the drag of the vertical tail is based on the square-foot surface area of the vertical tail as shown in a side view [Ref.9:p. 40]. The reference area of the fuselage is its frontal cross-sectional area in square feet. C_{D_O} is then the sum of the product of the component drags, $C_{D\pi}$, and their reference area, A_π, divided by the wing reference area as shown in the formula below.

$$C_{D_o} = \frac{\sum C_{D\pi} A_\pi}{S} \tag{7}$$

The value of k was based on the Oswald efficiency factor and aspect ratio as shown in the following equations:

$$k = \frac{C_L^2}{\pi e AR} \tag{8}$$

where:

$$e = \frac{1}{1+\delta} \tag{9}$$

The Oswald span efficiency factor, e, accounts for the non-elliptical lift distribution and flow separation about the wing [Ref 15:p. 296]. The value of 1+ δ was obtained from Figure 16 where δ and τ are correction factors for deviations from elliptical lift distribution [Ref. 16:p. 2-7]. The value of τ was not required.

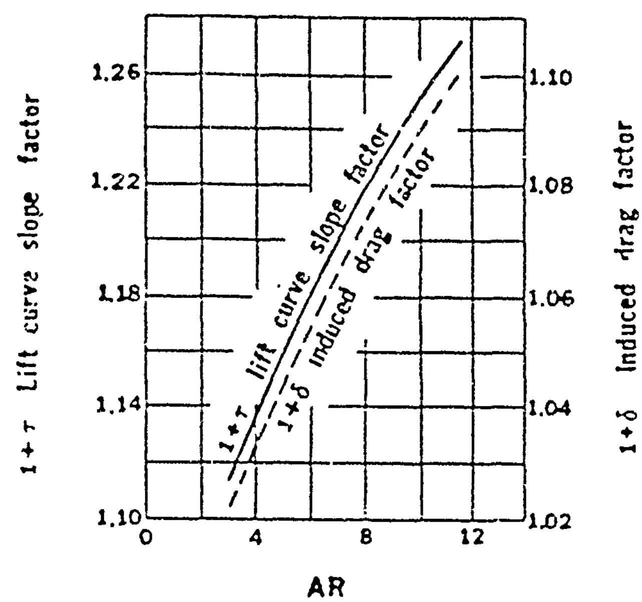


Figure 16. Variation of 1 + δ with AR [Ref. 9:p. 45]

For this study, aircraft dimensions were calculated as mentioned in the previous section on scaling factors and entered into a spreadsheet. Values of airfoil c_{dmin} were obtained from references 17-19. The value of c_{dmin} used was the minimum value for the airfoil found in the laminar "drag bucket." An example of c_{dmin} for several NACA 5-digit airfoils is shown in Figure 17, with values of c_{dmin} ranging from .0035 for the NACA 66(215)-416 to .010 for the NACA 64_2-415 with the addition of standard roughness.

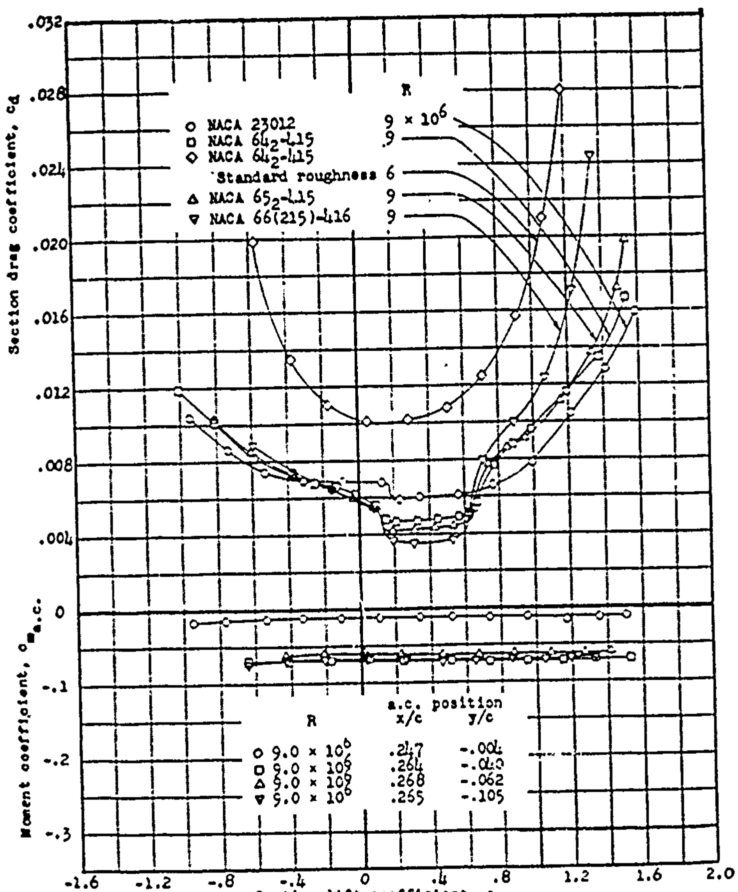


Figure 17. Sample NACA c_d vs c_l Chart [Ref. 19:p. 161]

In order to achieve flight test c_{dmin} as low as the values of NACA data for the airfoil, the wing surface requires a professional glass-like finish free of bugs, dirt and moisture. Post World War II NACA wind tunnel tests of manufactured wing sections were found to exhibit minimum drag coefficients on the order of 0.0070 to 0.0080 in nearly all cases of NACA 2-, 3-, and 6-series sections [Ref. 12:p.170]. However, those tests were conducted on wings of metal with numerous joints and rivets. Today's composite aircraft obtain exceptional finishes through the use of molds and seamless layups of fiberglass and graphite fabrics. Therefore, a minimum c_{dmin} comparison was justified. Fuselage, tail component and landing gear $C_{D\pi}$ values were taken from Smetana Tables 5-2, 5 3, and 5-4 [Ref. 9:p.42-44] which were compiled from Hoerner [Ref. 20]. For fuselages not listed in the Smetana tables, Nicolai [Ref. 16:p. 8-6] was utilized to estimate $C_{D\pi}$.

The spreadsheet calculations were tested for a known aircraft with a smooth glassy finish and are shown in Table 4. The $C_{D\pi}$ method resolved an equivalent flat plate drag area of 1.20 square feet with a "fullness factor" of 0.9. The fullness factor was this author's estimation of the percentage of the rectangular cross-sectional area filled by the frontal cross-section of the fuselage. A perfectly-round fuselage one foot in diameter only fills up 78.54% of a one foot square and thus has a fullness factor of .7854. A rectangular fuselage would have a factor of 1.0. The equivalent flat plate area is the reference area for a flat plate with a drag coefficient of 1, resulting in the same drag as the specified aircraft.

When the drag polar was entered into POINT with a power available curve for the 118 hp O-235 engine, the full-scale Vmax was predicted as 222 mph whereas the actual aircraft had an advertised Vmax of 225 mph. This error of less than 1.5% is considered optimistic considering the estimations made for propeller efficiency, fullness and interference drag, but nonetheless, quite acceptable. For this study, all drag polars were calculated using the minimum airfoil c_{dmin} values. Accordingly, the performance figures obtained were the theoretical "best" values and probably better than those that would be obtained from flight test of a typical aircraft. Figure 18 is a graph of the drag polar for the scaled conventional configuration with retractable gear.

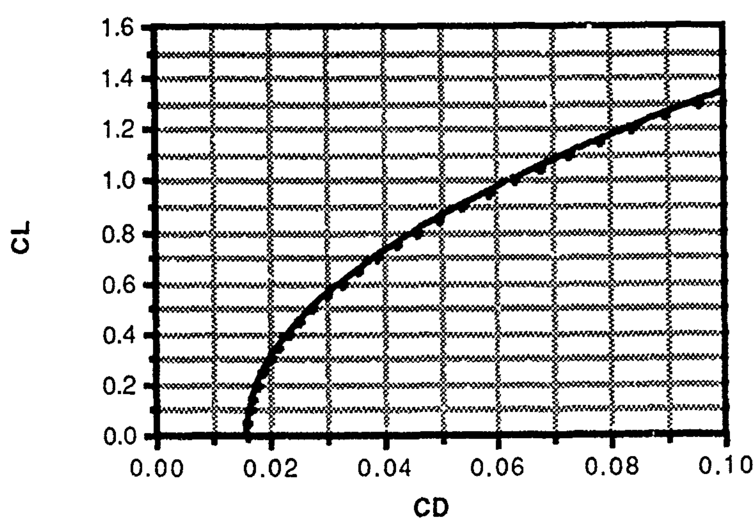


Figure 18. Conventional RG Drag Polar

As test cases, the POINT performance results were compared with results of the manufacturers' flight test for the Lancair 235 and the LongEze which both

use the Lycoming O-235 engine and have useful loads of approximately 600 pounds. Maximum speeds and ranges were within 1-2% of advertised performance, but predicted rates of climb were approximately 10% low. However, since all the configurations were tested with the same power available curve and the same program, the results will contain the same error margins and will not affect the comparative results.

Once Table 1 was tabulated, the aircraft scaled to the same useful load and the drag polars determined for each aircraft, then sufficient information was available to run the POINT performance program.

TABLE 4. DRAG POLAR ESTIMATION

DRAG POLAR BUILDUP

Aircraft **Conventional RG** *Input italicized data*

Wing from NASA tech paper 1865 for the NLF-0215F airfoil
Cdmin = *0.0045* S = *77.7*

Fuselage (ft.)

Length = *20.2* W/L = .183
Firewall height = *2.53* H/L = .125
Max height incl canopy = *3.3*
Max width = *3.7*
Firewall X-section (sq ft) = 9.4
adjust for roundness factor of: [illegible]
Adjusted X-section (sq ft) = 8.4
% for canopy = .30
Adjusted X-section (sq ft) = 11.0

from Smetana Table 5-2, cdπ = *.063*

Horizontal Tail from Smetana Table 5-3, cdπ = *0.0043*

Vertical Tail from Smetana Table 5-3, cdπ = *0.0043*

Component	Cdπ	Aπ	CdπAπ
Wing	.0045	77.7	0.35
Fuselage	.0630	11.0	0.69
Hor. tail	.0043	*10.8*	0.05
Vert. tail	.0043	*8.8*	0.04
Total			1.13

Interference effects *0.06*
Protuberance effects *0.03*

Equiv. flat plate area (sum of CdπAπ and effects) $\boxed{1.23}$

CDO = .01579 AR = *7.2*
e = *0.938* k1 = .0158
k3 = .0471

$\boxed{\text{CD} = .0158 + .0471\ \text{CL}^{\wedge}2}$

VII. POINT PERFORMANCE PROGRAM

The POINT performance program was used to predict the aircraft performance when under zero accelerations. This FORTRAN program was originally written for a mainframe computer by Dr. Frederick O. Smetana and published as Reference 21 in 1984. Four years later, the programs were adapted to run on IBM and finally Macintosh personal computers. The programs were not particularly user-friendly, but are reasonably usable by undergraduate students with basic concepts of aircraft performance. The following inputs were required and have been calculated in previous sections:

- Power available vs velocity curve
- Initial and final altitude
- Aircraft weight
- Wing Area
- Drag polar
- Coefficients of the drag polar

After the program was run, the following values were provided by the output in POINT.TXT:

- Maximum and minimum level fight speeds at altitude
- Speed for maximum climb angle and maximum climb angle at altitude
- Maximum endurance speed at altitude
- Speed for maximum range at altitude
- Service and absolute ceilings
- Maximum rate of climb, power and time schedule vs altitude for minimum time to climb from altitude H_i to altitude H_f
- Rate of climb, speed and power schedule vs altitude for most economical climb from H_i to H_f
- Maximum rate of climb, power available and power required schedule vs velocity at altitude H_i for velocities between the minimum and maximum level flight speeds
- Lift and drag coefficients in all of the above cases

The program was started by opening POINT.APL on the Macintosh. The following is an example of the prompts and input entries for the test case aircraft (Lancair 235). The values for the power available prompts were from Table 1 and the C_{Do} and k values were obtained from Table 4.

```
   enter 1 if an existing data set is to be processed
   enter 0 if a new data set is to be created
0
 IF YOU NEED HELP INTERPRETING THE FOLLOWING QUESTIONS
RESTART THE PROGRAM AND ENTER 1 AT THIS POINT. OTHERWISE,
ENTER 0.
0
   enter the number of points on the power available curve
5
   enter the power and velocity for point number    1
0,0
   enter the power and velocity for point number    2
22875,100
   enter the power and velocity for point number    3
45859,190
   enter the power and velocity for point number    4
54762,280
   enter the power and velocity for point number    5
28944,380
   if the engine is supercharged, enter 1. If not, enter 0
0
   enter the reference altitude
0
   enter the initial altitude
0
   enter the final altitude
10000
   enter the aircraft weight
 1400
   enter the wing area
76
   enter the three coefficients of the drag polar
.0158,0.0,.0471
   enter the exponent of the CDI2 term
2
```

Once the exponent of the CDI2 (induced drag due to lift) term was entered, the POINT program automatically executed. On the Macintosh SE 20, it took about 1 minute until a blinking cursor reappeared. POINT.APL was then exited by either closing or quitting. Table 5 is an example of the data file generated (POINT.DAT) which was modified for additional aircraft by highlighting the data to be changed and simply typing in the new data. This step avoided the requirement to reenter the power available curve data or altitude data.

TABLE 5. POINT.DAT

```
      5       0 0.0000000000000D+00
0.0000000000000D+00  0.0000000000000D+00
0.2287500000000D+05  0.1000000000000D+03
0.4585900000000D+05  0.1900000000000D+03
0.5476200000000D+05  0.2800000000000D+03
0.2894400000000D+05  0.3800000000000D+03
 0.1580000000000D-01  0.0000000000000D+00  0.4710000000000D-01
0.2000000000000D+01
0.7600000000000D+02
0.1400000000000D+04  0.0000000000000D+00  0.1000000000000D+05
```

The location of particular values could be determined by comparing POINT.DAT with the data input from POINT.APL. The performance output for the POINT program was listed by opening POINT.TXT. If a printout was desired, page setup was set to the horizontal position. A shortened version of the output is shown in Tables 6 and 7.

QUICKPLOT was used to generate plots of the columnar output by highlighting the desired numerical data, copying it, and then opening QUICKPLOT and pasting the data in. QUICKPLOT then asked which column

was to be plotted against what other column. If more than two columns were to be plotted, the paste function was done again and the additional columns to be plotted were selected. These values were plotted on the same graph as before. Figure 19 is an example of the power versus velocity curve plotted with data from Table 7.

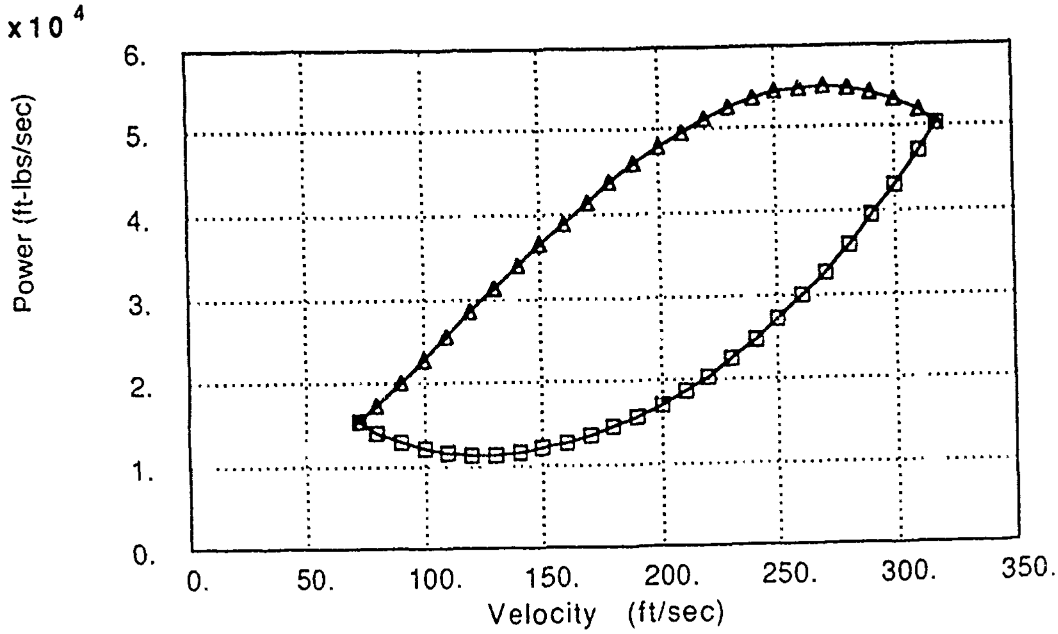


Figure 19. QUICKPLOT Power vs. Velocity

If any of POINT TXT was to be saved, it had to be saved under a new name as each run of POINT.APL would overwrite the previous data and output.

TABLE 6. POINT.TXT

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.15800D-01 + 0.00000D+00*CL**2 + 0.47100D-01*CL** 0.20000D+01
WING AREA = 0.77740D+02 SQ.FT   WEIGHT = 0.14483D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE
++

MINIMUM LEVEL FLIGHT SPEED = 0.72108D+02 FT/SEC
LIFT COEFFICIENT = 0.30109D+01   DRAG COEFFICIENT = 0.44278D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.31817D+03 FT/SEC
LIFT COEFFICIENT = 0.15465D+00   DRAG COEFFICIENT = 0.16926D-01

MAXIMUM CLIMB ANGLE = 0.65156D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.16442D+03 FT/SEC
LIFT COEFFICIENT = 0.57910D+00   DRAG COEFFICIENT = 0.31595D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.12492D+03 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.11398D+05 FT-LBS/SEC
LIFT COEFFICIENT = 0.10032D+01   DRAG COEFFICIENT = 0.63200D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.16441D+03 FT/SEC
LIFT COEFFICIENT = 0.57919D+00   DRAG COEFFICIENT = 0.31600D-01

SERVICE CEILING = 0.22450D+05 FT
VELOCITY AT SERVICE CEILING = 0.21878D+03 FT/SEC
LIFT COEFFICIENT = 0.66693D+00   DRAG COEFFICIENT = 0.36750D-01

ABSOLUTE CEILING = 0.24622D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.22100D+03 FT/SEC
LIFT COEFFICIENT = 0.70497D+00   DRAG COEFFICIENT = 0.39208D-01

TABLE 7. POINT.TXT

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD
0.00000D+00	0.21396D+02	0.20955D+03	0.49535D+05	0.35651D+00	0.21786D-01
0.50000D+03	0.20884D+02	0.20961D+03	0.48682D+05	0.36158D+00	0.21958D-01
0.10000D+04	0.20376D+02	0.20967D+03	0.47839D+05	0.36672D+00	0.22134D-01

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.74995D-09	0.15358D+05	0.15358D+05	0.72108D+02
0.23087D+01	0.17441D+05	0.14098D+05	0.80000D+02
0.49751D+01	0.20137D+05	0.12932D+05	0.90000D+02
0.74114D+01	0.22875D+05	0.12141D+05	0.10000D+03
0.96526D+01	0.25634D+05	0.11654D+05	0.11000D+03
0.11715D+02	0.28392D+05	0.11425D+05	0.12000D+03
0.13603D+02	0.31127D+05	0.11426D+05	0.13000D+03
0.15315D+02	0.33819D+05	0.11639D+05	0.14000D+03
0.16842D+02	0.36445D+05	0.12053D+05	0.15000D+03
0.18174D+02	0.38983D+05	0.12662D+05	0.16000D+03
0.19298D+02	0.41413D+05	0.13463D+05	0.17000D+03
0.20199D+02	0.43712D+05	0.14457D+05	0.18000D+03
0.20861D+02	0.45859D+05	0.15646D+05	0.19000D+03
0.21266D+02	0.47832D+05	0.17033D+05	0.20000D+03
0.21396D+02	0.49610D+05	0.18622D+05	0.21000D+03
0.21233D+02	0.51170D+05	0.20418D+05	0.22000D+03
0.20759D+02	0.52492D+05	0.22427D+05	0.23000D+03
0.19953D+02	0.53554D+05	0.24656D+05	0.24000D+03
0.18796D+02	0.54333D+05	0.27110D+05	0.25000D+03
0.17269D+02	0.54809D+05	0.29798D+05	0.26000D+03
0.15351D+02	0.54959D+05	0.32725D+05	0.27000D+03
0.13023D+02	0.54762D+05	0.35901D+05	0.28000D+03
0.10264D+02	0.54196D+05	0.39331D+05	0.29000D+03
0.70537D+01	0.53241D+05	0.43025D+05	0.30000D+03
0.33717D+01	0.51873D+05	0.46990D+05	0.31000D+03
0.14659D-09	0.50435D+05	0.50435D+05	0.31817D+03

VIII. PERFORMANCE EQUATIONS

The output of the POINT performance program was utilized to make additional performance predictions by use of a spreadsheet. The Breguet range and endurance formulas as presented in Anderson [Ref. 22] were used to calculate maximum range and endurance as shown:

$$R = \frac{\eta}{c}\frac{C_L}{C_D}\ln\frac{W_{TO}}{W_L} \tag{10}$$

$$E = \frac{\eta}{c}\frac{C_L^{\frac{3}{2}}}{C_D}(2\rho_\infty S)^{\frac{1}{2}}\left(W_L^{-\frac{1}{2}} - W_{TO}^{-\frac{1}{2}}\right) \tag{11}$$

The POINT performance program computed the velocities for the maximum range and endurance and the corresponding values of C_L and C_D were obtained from POINT.TXT. However,the equations in the program do not account for the decreased aircraft weight as the fuel is burned off. For a general aviation aircraft or RPV where the fuel weight is only 10-15% of the gross weight, the velocity change is minor. The value of η was obtained from Table 1 for the respective velocities. The landing weight was the takeoff weight minus the weight of the 32 gallons of fuel onboard. The range and endurance values calculated were for a no-reserve fuel condition.

The specific fuel consumption of the Lycoming O-235 is shown in Figure 20. This value may be found in the owner's manual for different aircraft engines.

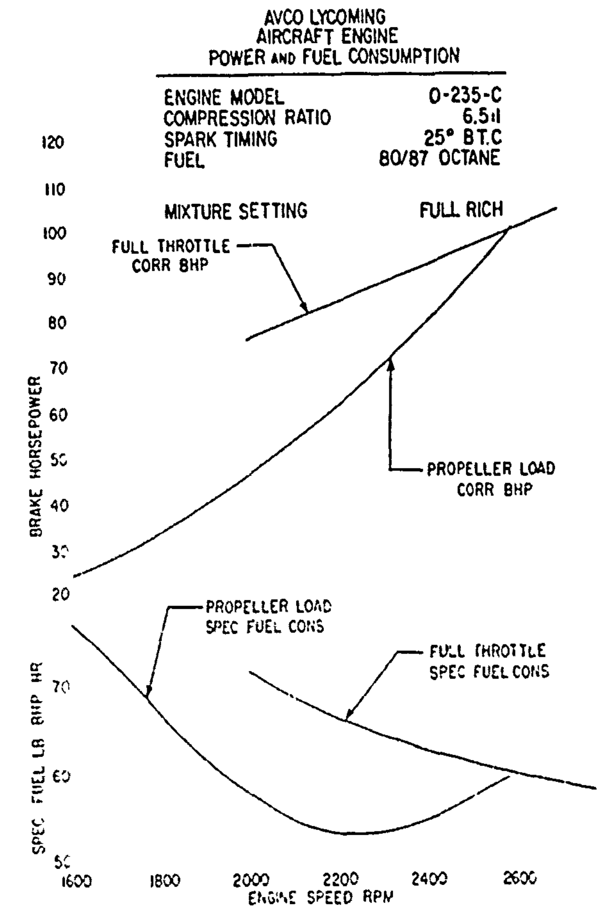


Figure 20. [Ref. 23]

The value of .59 lbs of fuel per BHP per hour was converted into consistent units for the Breguet equations as shown below:

$$c = \frac{0.59\ lb_f}{BHP \cdot hr} \frac{BHP}{550\ \frac{ft\ lb}{sec}} \frac{hr}{3600\ sec} = 2.98 \times 10^{-7} \frac{lb_f}{\frac{ft\ lb}{sec}\ sec} \tag{12}$$

Table 8 is an example of the spreadsheet for maximum range and endurance based on the Breguet formulas.

TABLE 8. RANGE AND ENDURANCE SPREADSHEET

Input italized data

Givens:

Gal fuel =	32
Fuel weight (lbs) =	192
SFC =	0.59
Wo =	*1448*
W1 =	1256

Range

From POINT.TXT:

Vmax range (ft/sec) =	*164.41*
Vmax range (mph) =	111.8
CL =	*0.57919*
CD =	*0.0316*

From Table 1:

η =	*0.633*
c =	2.9798E-07
CL/CD =	18.33

Range (sm) =	1049

Endurance

From POINT.TXT:

Vmax endur (ft/sec) =	*124.92*
Vmax endur (mph) =	85.0
CL =	*1.0032*
CD =	*0.0632*

ρ = .00238

S = *77.74*

From Table 1:

η =	*0.47*
c =	2.9798E-07
CL/CD =	15.87

Endur (hr) =	8.2

IX. PERFORMANCE COMPARISONS

Following the procedures in the previous chapters, the performance of each of the nine different configurations was computed. For all configurations, the useful load was held constant at 600 pounds and the actual aircraft was scaled dimensionally. Due to different designs, methods of construction and materials, the gross weights varied from 1027 lbs for the joined-wing aircraft up to 1575 for the 3-surface aircraft. The aircraft that were scaled up from single-seaters tended to be lighter than the 2-seaters by 100-300 lbs although there was some overlap as shown in Figure 21.

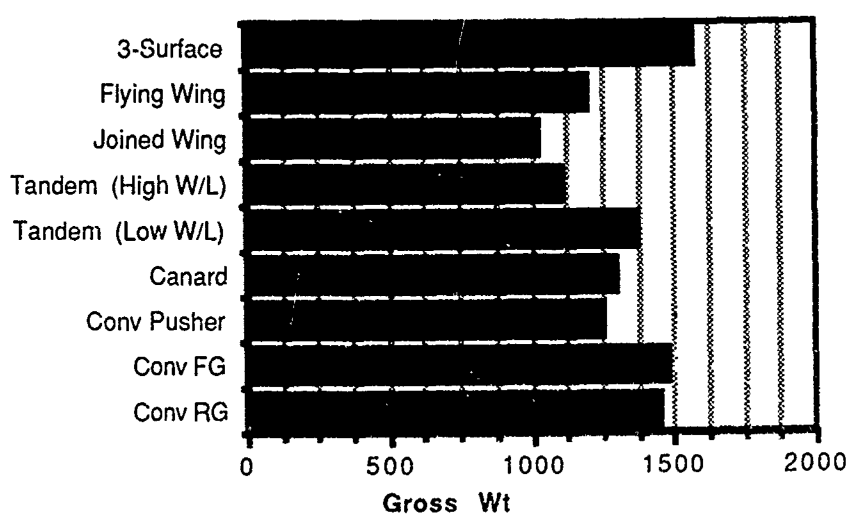


Figure 21. Gross Weight Comparison

The lightest aircraft, based on the Ligeti Stratos, made use of carbon fiber in the spars, Kevlar® for shear webs and cockpit and fiberglass wings [Ref. 23:p.49]. The cockpit seating was also very tight and significantly reclined as can be seen in Figure 22.

Figure 22. Joined-Wing Aircraft Cockpit [Ref. 23]

The flying wing and conventional pusher aircraft were also scaled up from single-seaters and featured reclined seating. The conventional pusher was fabricated of fiberglass and sheet metal [Ref. 24] whereas the flying wing fuselage was constructed of Dacron®-covered steel tubing with a wooden-ribbed wing covered in Mylar® [Ref. 25]. The tandem-wing (high W/L) was a 2-seater constructed of fiberglass which had a scaled gross weight of only 1109 pounds. A similar design with low wing loading weighed in at 1372 pounds, the greater weight of which was accounted for in part by its larger scaled wing area of 112 square feet to the scaled tandem's (high W/L) wing area of 67 square feet. As expected, the lighter aircraft performance was significantly better in the range, endurance and rate of climb comparisons. The relationship between the configuration and its weight could be examined more closely in the future.

The maximum velocity of the scaled aircraft was inversely proportional to the equivalent flat plate area as shown in Figures 23 and 24.

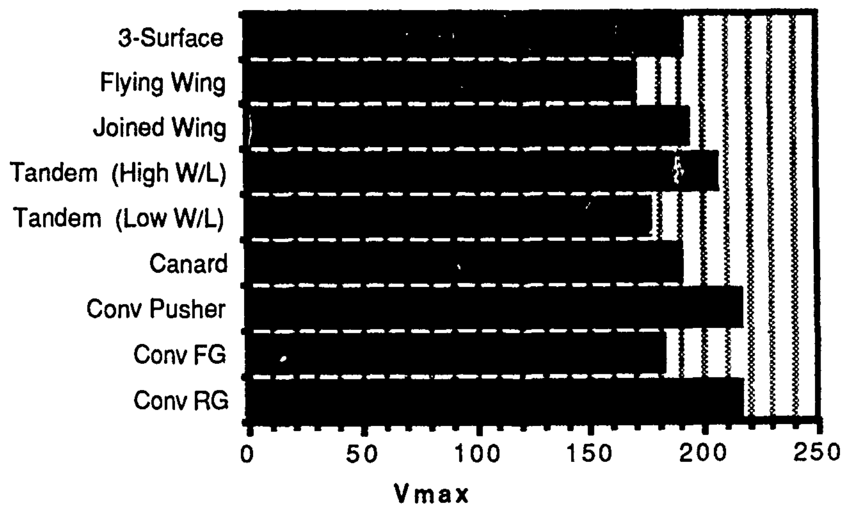


Figure 23. Vmax

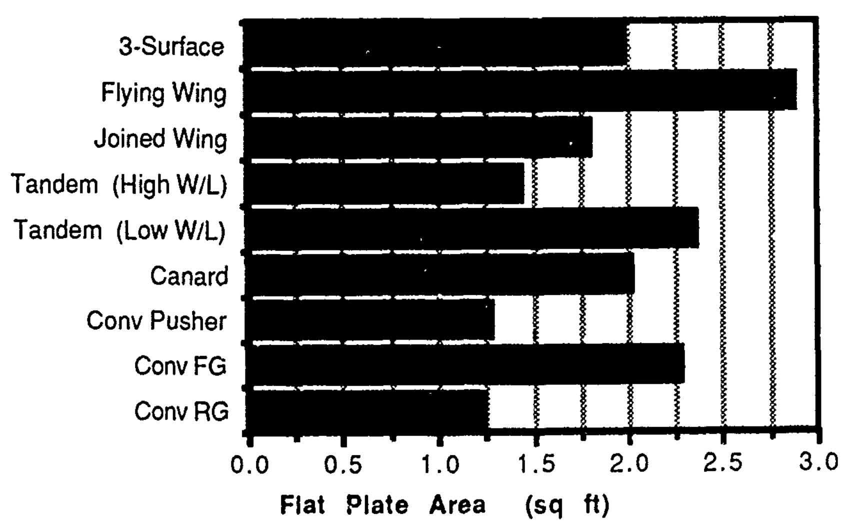


Figure 24. Equivalent Flat Plate Area

Not surprisingly, the two retractable gear aircraft had the top speed of 216 mph. However, the tandem-wing (high W/L) and joined-wing, with their closely-faired and short fixed gear either on the wingtips or on the centerline, were close behind at 205 and 193 mph respectively. The canard, 3-surface and conventional FG all had exposed, but faired, main gear with either a retractable nosewheel or a small tailwheel and topped out between 180-190 mph.

Some of the most significant differences could be seen in the range and endurance comparisons where L/D, η and Wto were primary determinants. The range comparison is shown in Figure 25.

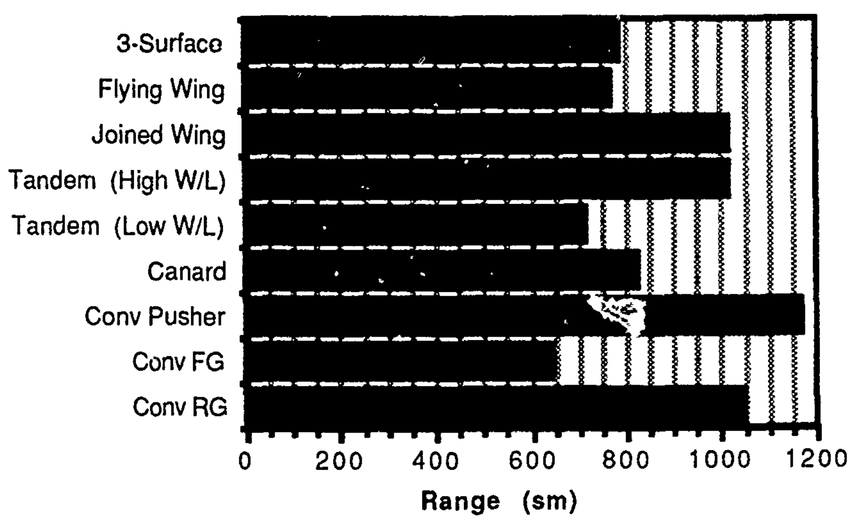


Figure 25. Range Comparison

The conventional pusher configuration flew 1169 sm before fuel exhaustion, a full 11-15% improvement over its closest competitors at 1013-1049 sm. The conventional pusher had neither the lowest equivalent flat plate drag area nor the lowest gross weight, but the combination of reasonably low values of the above

parameters along with a higher η due to its higher efficiency cruise speed put it well in front. The aircraft with the highest equivalent flat plate areas did the poorest in range. Their high parasite drag coefficient coupled with low propeller efficiency at lower speeds gave them maximum ranges that were only 60-70% of the conventional pusher. Although the flying wing had an $(L/D)_{max}$ of 21.6 to the conventional pusher's $(L/D)_{max}$ of 20.2, the flying wing's propeller efficiency was only .32 compared to the conventional pusher's η of .55.

The comparative endurance chart is shown in Figure 26.

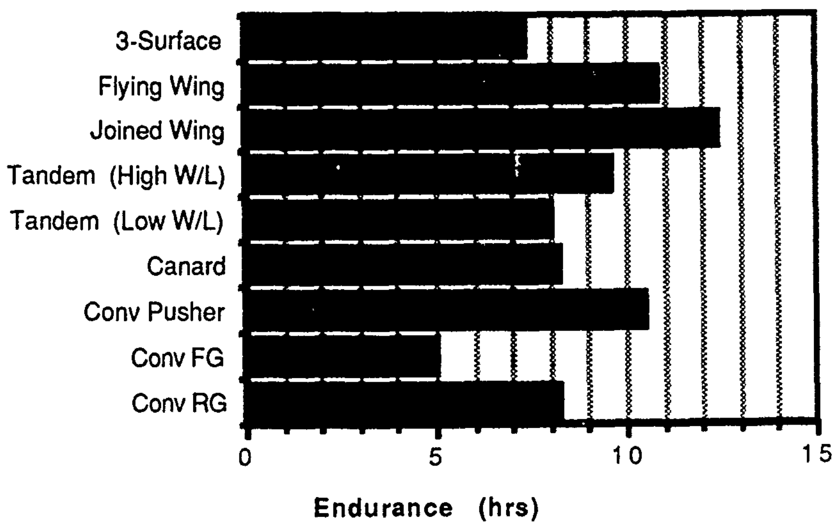


Figure 26. Maximum Endurance comparison.

In this case, the maximum value of $(L/D)^{3/2}$ was the dominant parameter in the endurance equation and favored the configurations with the larger wing areas such as the flying wing (216 ft^2), and the joined-wing (148 ft^2). Although the conventional pusher had only 86 square feet of wing area, its higher η of .40

exceeded the previous aircraft η values of .23 and .27. Once again, the fixed gear aircraft faired poorly by comparison.

The rate of climb comparison was a measure of excess power available over power required. Since all the configurations had the same power available, the dominant parameters were weight and the coefficients of lift and drag as shown in the following equation:

$$P_R = \sqrt{\frac{2W^3C_D^{\,2}}{\rho_\infty S C_L^{\,3}}} \propto \frac{1}{C_L^{3/2}/C_D} \qquad (13)$$

The joined-wing and tandem-wing (high W/L) both topped 1600 feet per minute with the conventional pusher close behind at 1560 feet per minute as shown in Figure 27.

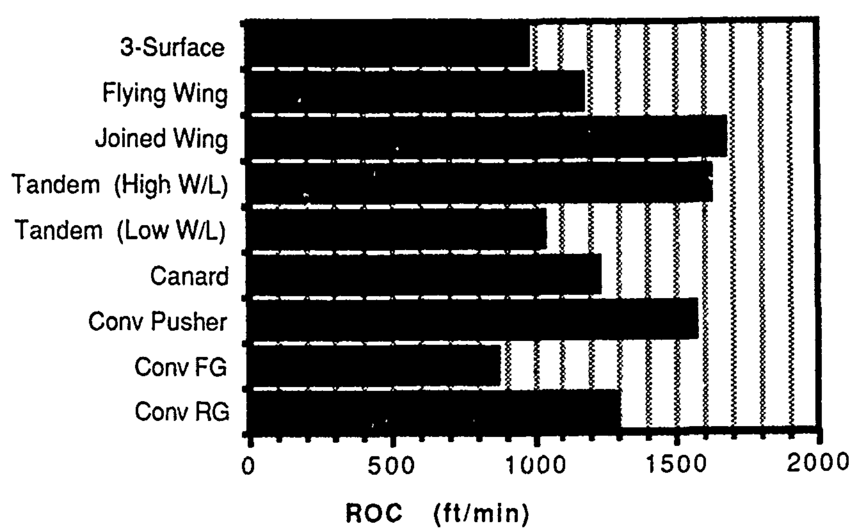


Figure 27. Rate of Climb comparison.

The service ceiling, defined as the altitude at which the aircraft can still maintain a 100 foot per minute rate of climb, was also a function of excess power available and the rankings between the aircraft were comparable to the sea level rate of climb. Since all of the aircraft easily exceeded 14,000 feet, or the altitude where the pilot must wear an oxygen mask and the engines are normally turbocharged, these comparisons were not judged significant.

The numerical values between the scaled aircraft are shown in Table 9.

TABLE 9. PERFORMANCE RESULTS

Configuration	Flat Plate Area (sq ft)	Gross Wt (lbs)	Vmax (mph)
Conventional RG	1.23	1448	216
Conventional FG	2.28	1475	181
Conventional pusher	1.28	1250	216
Canard	2.02	1293	190
Tandem (Low W/L)	2.36	1372	176
Tandem (High W/L)	1.44	1109	205
Joined wing	1.80	1027	193
Flying wing	2.87	1200	170
3-Surface	1.97	1575	191

Configuration	Range (sm)	Endurance (hrs)	ROC (ft/min)
Conventional RG	1049	8.2	1290
Conventional FG	641	5.0	870
Conventional pusher	1169	10.5	1560
Canard	821	8.2	1220
Tandem (Low W/L)	711	8.0	1034
Tandem (High W/L)	1013	9.6	1614
Joined wing	1015	12.4	1674
Flying wing	766	10.8	1170
3-Surface	785	7.3	980

Configuration	Service ceiling (ft)	Vx (mph)	Vy (mph)
Conventional RG	22,450	112	143
Conventional FG	16,587	102	122
Conventional pusher	26,878	103	140
Canard	23,588	94	124
Tandem (Low W/L)	23,554	85	114
Tandem (High W/L)	26,000	99	133
Joined wing	30,306	84	123
Flying wing	28,026	74	108
3-Surface	21,460	98	125

X. STABILITY DERIVATIVES

Although aircraft performance may be the driving factor behind its design, no design is complete without an estimate of the aircraft stability derivatives and its handling characteristics. These dimensionless values are not necessarily comparative as far as "best and worst," but the values should represent acceptable handling qualities, which can be compared to those for certified aircraft in its class.

The lattice vortex program, LINAIR, was used to estimate the some of the stability characteristics of the configurations. LINAIR can also determine the appropriate wing twist for a new design, expected performance for a given wing geometry and proper angles of incidence for tail and canard surfaces in the hands of an experienced designer [Ref. 10:p.2]. LINAIR calculates the aerodynamic characteristics of multi-element nonplanar lifting surfaces by solving the Prandtl-Glauert equation for inviscid, irrotational, subsonic flow. Once LINAIR has solved the system of equations, it utilizes the Kutta-Joukowski relation to compute the forces and moments acting on the configuration. LINAIR is a simplified version of the lattice vortex program demonstrated in AE 3501 (Advanced Aerodynamics).

As a test of this program's usefulness, the lift and moment coefficients (C_L, C_M) were computed over an angle of attack, α , ranging from $0°$ to $8°$ and the rolling and yawing coefficients (C_l and C_n) were computed using a sideslip angle, β, of 2°. Each aircraft was plotted in x-y-z coordinates according to the LINAIR instructions and the displayed geometry was checked for correct representation. Full-scale blueprints proved advantageous over page-sized

drawings for plotting coordinates. Most of the configurations were plotted from enlarged photocopies or sketches and the fuselages may have not been modeled with enough detail for dependable side forces. Special care had to also be used in panel distribution in order to prevent downstream vortices from crossing over control points of other surfaces and producing erroneous results [Ref. 10:p. 29].

The value of $C_{L\alpha}$ was determined by calculating ΔC_L, dividing by $\Delta\alpha$ and converting to radians as shown below:

$$C_{L_\alpha} = \frac{\Delta C_L}{\Delta \alpha} \times \frac{180}{\pi} \tag{14}$$

Obtaining the value of C_M was a little more involved as $C_{M\alpha}$ was calculated about the nose of the aircraft when X_{ref} was set to zero. This value of $C_{M\alpha o}$ was corrected to the c.g. of the aircraft, where $C_{M\alpha}$ is normally measured, by the relationship:

$$C_{M_{\alpha(c.g.)}} = C_{M_{\alpha o}} + C_{L_\alpha} \frac{\Delta X_{c.g.}}{c_{ref}} \tag{15}$$

where:

$$C_{M_{\alpha o}} = \frac{\Delta C_M}{\Delta \alpha} \tag{16}$$

and:

$$c_{ref} = \frac{S_{ref}}{b_{ref}} \tag{17}$$

If the location of the c.g. of the aircraft was known beforehand, then the value could be entered into LINAIR as X_{ref} and the computer would complete the above steps. A center of gravity location can be approximated by estimating

the aerodynamic center of the aircraft (the neutral point) and placing the c.g. 3-6 inches in front of X_{ac} [Ref. p. 60-62]. The actual c.g. must be more accurately calculated before the advanced design stage.

The values of $C_{n\beta}$ and $C_{l\beta}$ were calculated for a $\Delta\beta$ of 2°. This increment was sufficient because of the linear assumption. To translate $C_{n\beta}$ to the c.g. of the aircraft, the following relationship was used:

$$C_{n_{\beta(c.g.)}} = C_{n_{\beta 0}} + C_{Y_\beta} \frac{\Delta X_{c.g.}}{b_{ref}} \tag{18}$$

where:

$$C_{n_{\beta 0}} = \frac{\Delta C_n}{\Delta \beta} \tag{19}$$

and:

$$C_{Y_\beta} = \frac{\Delta C_Y}{\Delta \beta} \tag{20}$$

Once again, knowing the c.g. position prevented this moment transfer from having to be performed.

The static margin was determined by the following formula:

$$s.m. = -\frac{C_{M_\alpha}}{C_{L_\alpha}} \tag{21}$$

since:

$$C_{M_\alpha} = C_{L_\alpha}(h - h_n) \tag{22}$$

where:

$$h_n - h = s.m. \tag{23}$$

The configuration models as used are shown in Figures 28-33 and the calculated values for $C_{L\alpha}$, $C_{M\alpha}$, $C_{l\beta}$, and $C_{n\beta}$ are shown in Table 10.

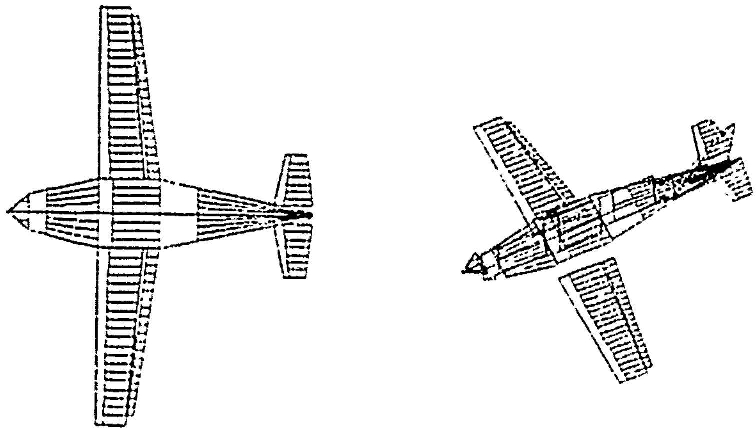


Figure 28. Model of Conventional Configuration

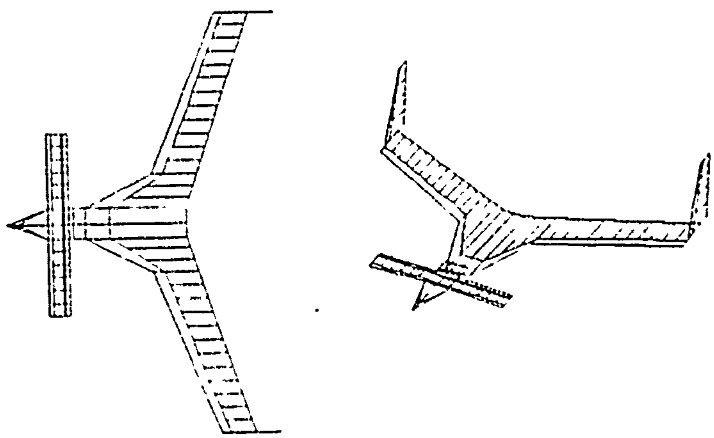


Figure 29. Model of Canard Configuration

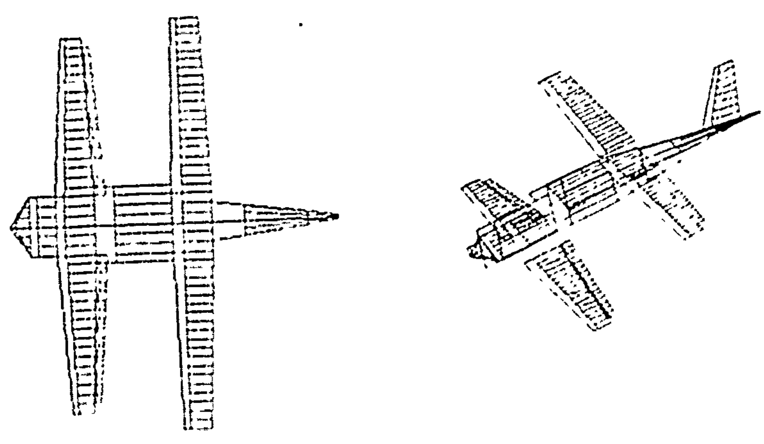


Figure 30. Model of Tandem-Wing Configuration

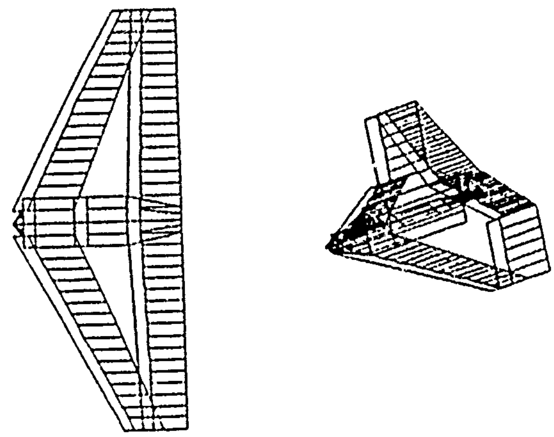


Figure 31. Model of Joined-Wing Configuration

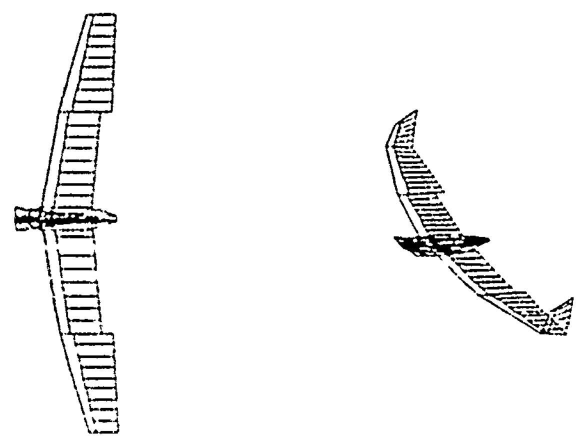


Figure 32. Model of Flying-Wing Configuration

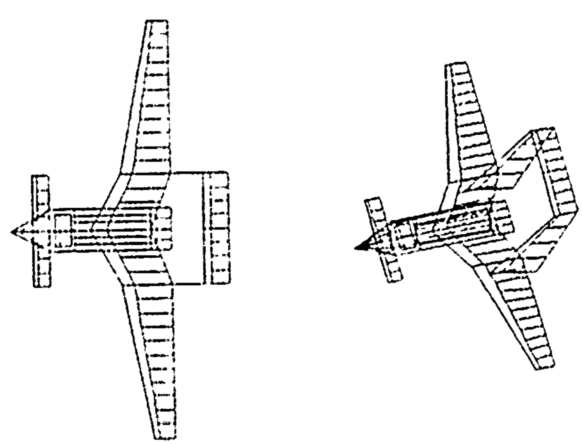


Figure 33. Model of 3-Surface Configuration

TABLE 10. STABILITY DERIVATIVES (per radian)

	$C_{L\alpha}$	$C_{M\alpha}$	$C_{n\beta}$	$C_{l\beta}$	S.M.
Conven	3.60	-.693	.264	-.102	.19
Tandem	3.75	-.921	.096	-.011	.25
Canard	5.35	-1.20	.175	-.016	.22
Joined	4.00	-.368	.265	-.007	.09
Wing	4.77	-2.60	.013	-.094	.54
3-Surf	5.06	-.594	.080	-.021	.12

Some static stability parameters of a variety of aircraft for comparison are shown in Table 11.

TABLE 11. STABILITY DERIVATIVES OF VARIOUS AIRCRAFT

[Ref. 27:p. 28]

(per radian)	Cessna 172	Twin Engine	Jet Trainer	Lear Jet	F - 4 Fighter	Boeing 747
C_{m_α}	-.89	-2.08	-.6	-.66	-.098	-1.45
C_{L_α}	4.6	6.24	5.0	5.04	2.8	5.67
$C_{L_{\delta e}}$	.43	.58	.39	.4	.24	.36
$C_{D_{\delta e}}$	.06	0.0	0.0	0.0	-.14	0.0
$C_{m_{\delta e}}$	-1.28	-1.9	-.9	-.98	-.322	-1.4
C_{l_β}	-.089	-.13	-.14	-.173	-.156	-.281
$C_{l_{\delta r}}$	.0147	.0087	.03	.014	.0009	0.0
C_{n_β}	.065	.12	.16	.15	.199	.184
$C_{n_{\delta r}}$	-.0657	-.0763	-.11	-.074	-.072	-.113
C_{Y_β}	-.31	-.59	-.94	-.73	-.655	-1.08
$C_{Y_{\delta r}}$	.187	.144	.26	.14	.124	.179

As can be seen above, the lift-curve slope, $C_{L\alpha}$, for all the configurations is within the range for general aviation aircraft. The pitching moment due angle of attack, $C_{M\alpha}$, should be negative in order to be longitudinally stable. Individual components, such as the wing or fuselage alone, may be positive, but the horizontal tail must make the entire aircraft negative. Once again, the values for all the configurations were in the expected range. The yawing coefficient due to sideslip angle, $C_{n\beta}$, is an indication of direction stability and is sometimes referred to as "weathercock stability." For a stable aircraft, $C_{n\beta}$ must be positive. The restoring value increases as the value of $C_{n\beta}$ increases. These values appeared reasonable for all the configurations. The final stability derivative examined is the rolling moment due to sideslip angle, $C_{l\beta}$, which is sometimes known as the dihedral effect and is normally negative.[Ref.28:C. 2] The joined-wing, tandem-wing and the canard configuration had low values of $C_{l\beta}$ which may be due to less than adequate modeling of the wings by creating too much anhedral or too little dihedral of the wings and canards. These configurations should be studied more closely for sensitivity to panel discretization and for improvement of dihedral effect. The static margin, s.m., is a measure of the longitundinal static stability of the aircraft with respect to incidence disturbances and is the distance between the c.g. and the neutral point in units of the mean aerodynamic chord. LINAIR uses the average chord, c_{ref}, rather than the mean aerodynamic chord, mac. In these test cases, the static margin was a function of where the c.g. was estimated to be.

XI. CONCLUSIONS AND RECOMMENDATIONS

The results from the microcomputer programs used in the study indicate a degree of accuracy sufficient enough to be used in preliminary aircraft design and for academic instruction. To ensure full use of the programs' capabilities however, a one week short course integrated into a beginning level aeronautical engineering course should be offered. Self-instruction from the user manual is possible, but very time-consuming and error prone. Perhaps an updated version of the programs could be written to make them more "user-friendly." The POINT performance program is one of several microcomputer programs utilized extensively in Aeronautical Engineering courses at the University of North Carolina, and LINAIR is used by students at Stanford University. These two programs, since they are operable with both Macintosh and IBM-compatible PC's, should be useful in the following Aeronautical Engineering courses:

<u>POINT</u>

AE 2035 (Basic Aerodynamics)

AE 2036 (Performance, Stability and Control)

AE 4323 (Flight Evaluation Techniques)

AE 4273 (Aircraft Design)

<u>LINAIR</u>

AE 2035 (Basic Aerodynamics)

AE 3501 (Advanced Aerodynamics)

AE 4273 (Aircraft Design)

Since the student may purchase the performance programs for the price of a textbook, they would have use of the programs after graduation. LINAIR is more expensive, about $1000, and probably would be limited to academic departments and actual aircraft designers. Granted, more elaborate software exists for the VAX® and mainframe computers which are more accurate and operate considerably faster, but these programs are significantly more complex to learn and use, and are well out of the price range of students.

For aircraft design, step one has usually been selection of a particular configuration, followed by numerous iterations to estimate the aircraft weight, wing area, power required, etc. Because of the amount of calculations that had to be done, all too often the conventional configuration has been the only configuration examined. Through the use of simple PC-based programs such as POINT and LINAIR, rough performance estimates and handling characteristics can be computed, allowing other configurations to be examined as well. Following the procedures in this study, the aircraft designer can quickly do tradeoff studies between propeller pitch, horsepower, weight and equivalent flat plate drag area and their effect on performance. The use of scaling factors enables quick comparisons to presently flying aircraft in the design's category.

What Burt Rutan has done for the canard configuration and Molt Taylor with the conventional pusher, has yet to be achieved for the 3-surface, tandem or joined-wing configuration, although several successful designs are in the air now. These three configurations share the attribute of stall-proof approaches when properly designed. The 3-surface Avanti has proven itself as a true contender in the business class turboprop category as shown in the *Aviation Week and Space Technology* comparison. Scaled Composites, Inc., has tested the concept of a

cross between the 3-surface and tandem-wing configuration with its AT^3 shown in Figure 28 [Ref 29:p. 45].

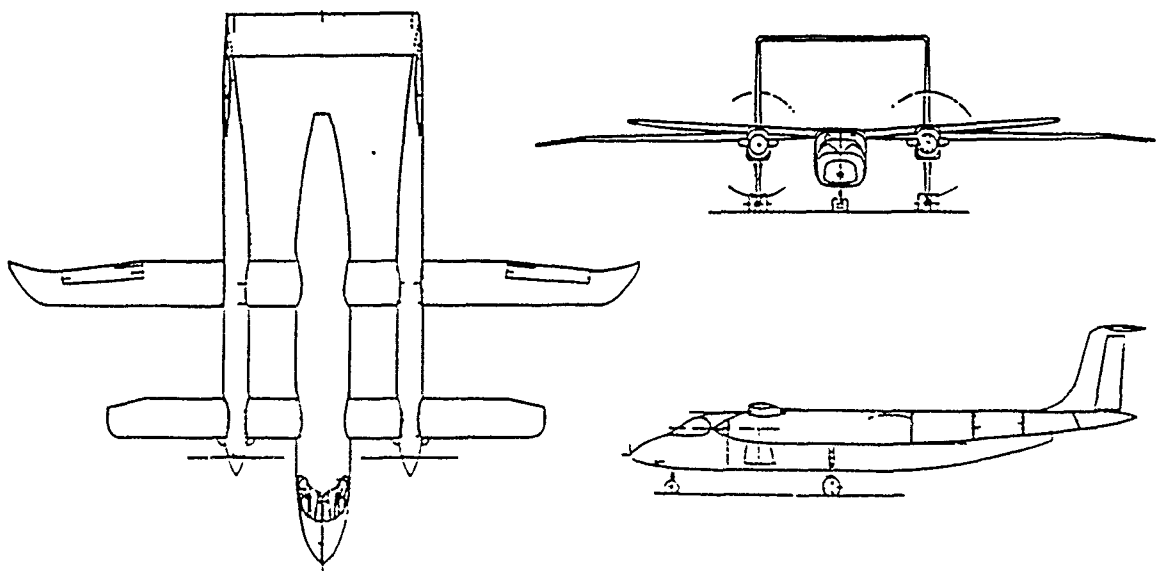


Figure 28. Advanced Technology Tactical Transport AT^3 [Ref 29:p. 45]

When one looks at the overall results of this thesis, the configuration with the most potential, and granted, the most risks, is the joined-wing configuration. In the comparative performance studies, the joined-wing was the lightest, had the greatest endurance and rate of climb and was competitive in every other category. Since compromise is a necessary part of aircraft design, an aircraft that does well in every category allows the greatest flexibility for optimization. The joined-wing design allows for lighter wings and spars because of its structural sturdiness, and the fuselage can be designed into the optimal elongated egg shape for minimal fuselage drag at low Mach numbers. A thumbnail sketch of a possible joined-wing/3-surface configuration is shown in Figure 29.

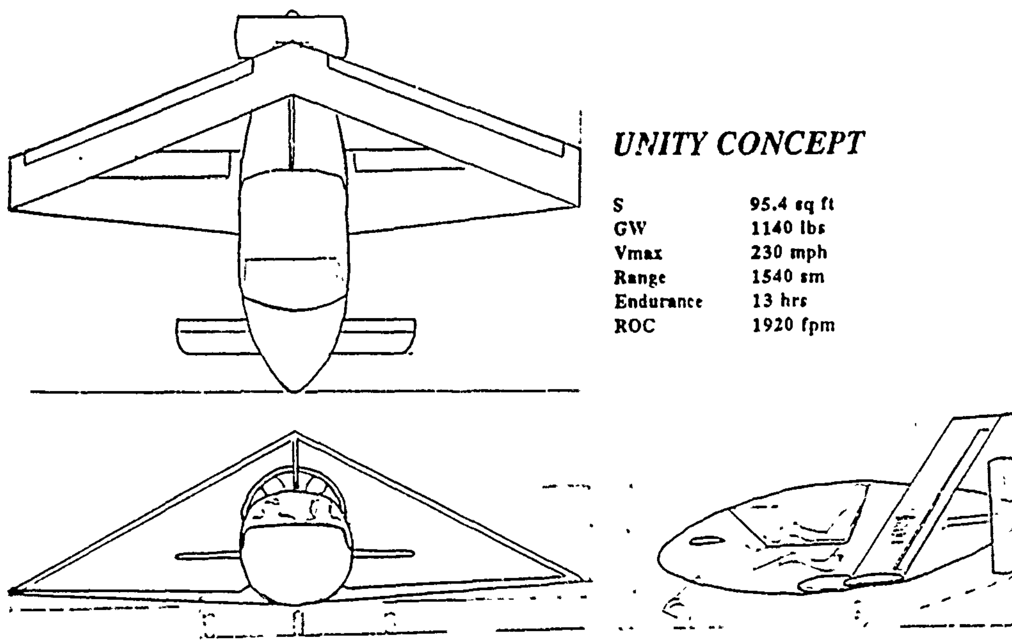


Figure 29. Possible Joined-wing/3-surface Configuration

Presently, current work on joined-wing configurations is being done by Julian Wolkovitch who claims reduced transonic and supersonic drag and improved area ruling on high Mach aircraft as well [Ref. 8:p. 85].

Follow-on studies to this thesis could proceed in two directions. First is the theoretical direction, which would use the same criteria of this thesis, but parallel that of the Selberg and Rokhsaz study by designing the configurations with the same generic fuselage, main wing and gross weight. This study would require extra attention in the area of static stability and handling qualities, since the theoretical designs would unproven in actual flight test. The other course would

be along the lines of flight test, in that the student would input the weight, power and dimensional data for an actual experimental aircraft and run the programs for their predictions of flight performance. The student then could arrange to fly with the pilot and collect flight performance data as learned in AE 4323 and do comparisons with the computer output to test the accuracy of the programs. Of the nine configurations studied in this thesis, at least six can be found nearby in California with a high likelihood of owner cooperation. The model for the joined-wing aircraft has been successfully flown in Australia and a two-place version may be in the works, but flight test of this and the flying wing model is unlikely due to lack of availability.

The challenges in this thesis were finding that little information could be found in aircraft design books relating to non-conventional configurations. How is the aspect ratio defined for a tandem-wing or joined-wing aircraft? How does one define the wing reference area for 3-surface aircraft? Tables of drag indexes often overlooked V-tails, rudderons and winglets. Some of the airfoils had a reflex flap which deflected upward slightly in cruise for decreased induced drag, but airfoil wind tunnel data did not account for it. In short, more research data needs to be tabulated for unconventional configurations. Along these lines, the U.S. Naval Research Laboratory is developing the Low Altitude/Airspeed Unmanned Research Aircraft (LAURA). This RPV consists of a modular baseline fuselage designed to accept a variety of wings with low Reynolds number (LRN) airfoils and various combinations of tails. LAURA is designed for the shipboard environment which includes stowage, launch and recovery parameters, salt water intrusion, and an assortment of autonomous maritime

missions. Some of the possible configurations include the hinged-wing, tandem-wing, twin boom and joined-wing as shown in Figure 30. [Ref. 30]

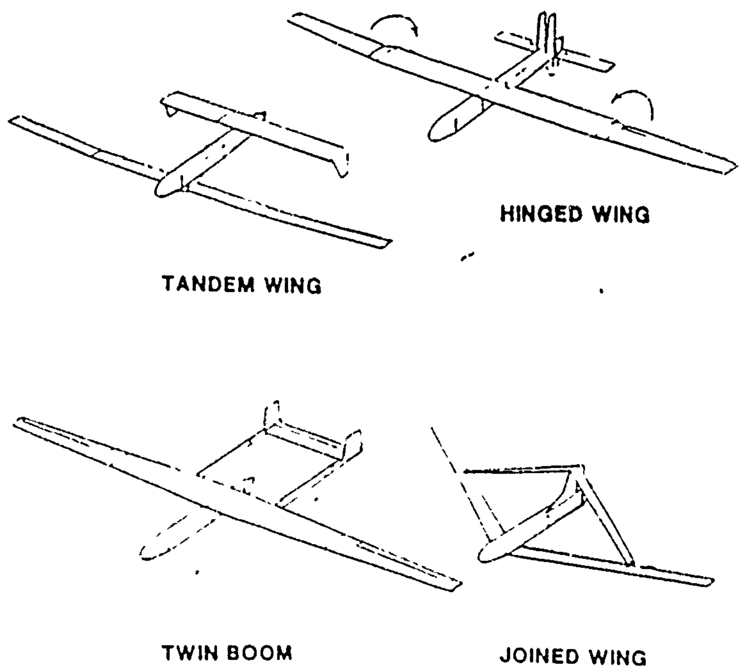


Figure 29. LAURA Configuraiions [Ref. 30]

In conclusion, the conventional configuration is hard to surpass for simplicity and low risk, but for specialized mission performance and optimal performance, all configurations should at least be considered.

APPENDIX A MODEL FOR THE CONVENTIONAL TRACTOR CONFIGURATION WITH RETRACTABLE GEAR

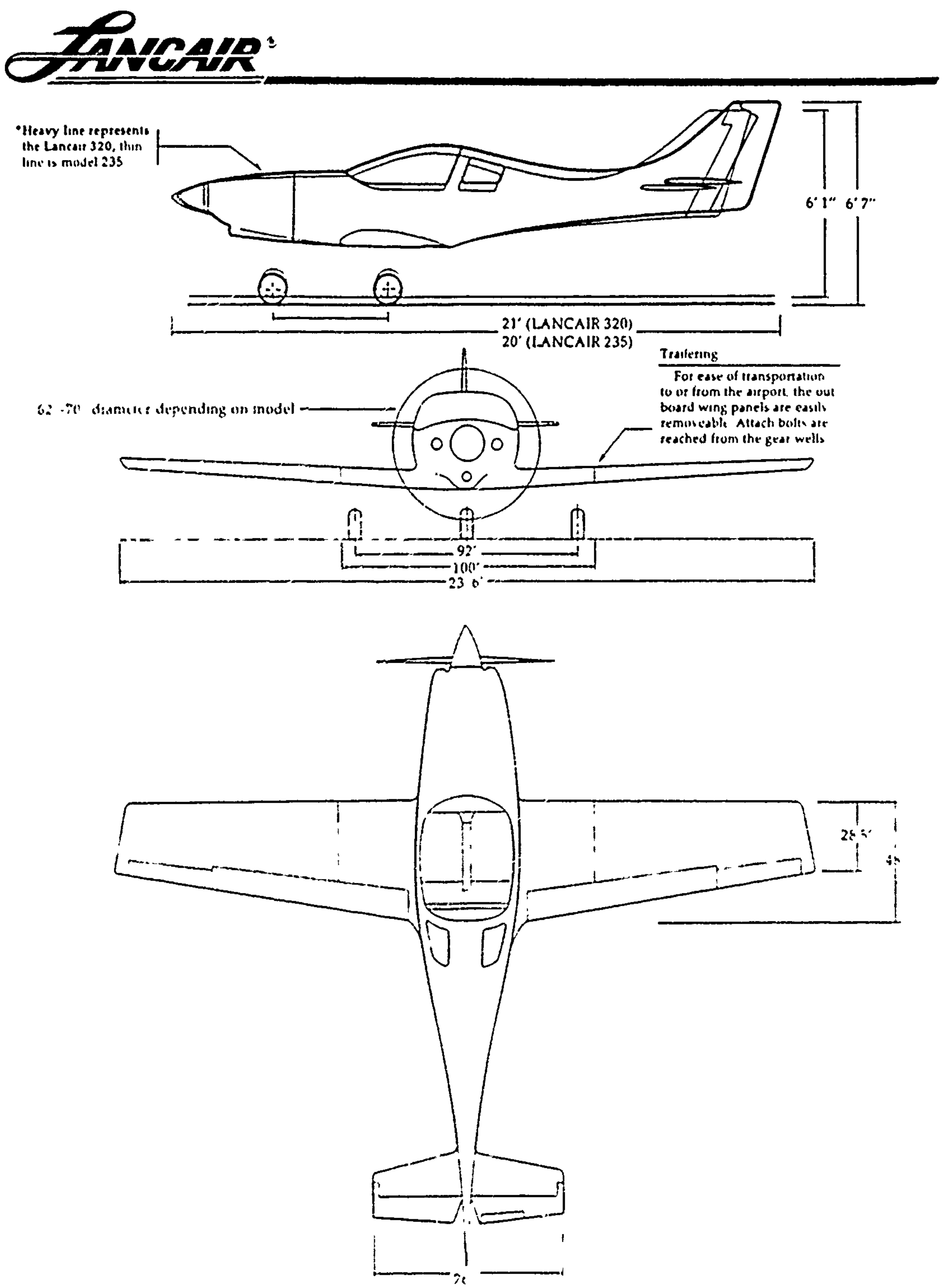


TABLE A.1. SCALED PARAMETERS BASED ON λ FOR THE CONVENTIONAL RG

Conventional retractable tractor aircraft based on the Lancair 235

Aircraft Specifications	Actual values	Scaled values ($\lambda = \mathbf{0.99}$)
Wing span (ft.)	23.40	23.67
Wing chord root	4.33	4.38
Wing chord tip	2.38	2.40
Wing area (sq. ft.)	76.00	77.74
Wing airfoil section	NLF-0215-F	NLF-0215-F
Wing aspect ratio	7.20	7.20
Wing loading (GW) (lb./sq. ft.)	18.42	18.63
Effective horizonal tail span (ft.)	6.67	6.74
Horizonal tail chord root	2.17	2.19
Horizonal tail chord tip	1.58	1.60
Horizonal tail area (sq. ft.)	10.60	10.84
Horizonal tail airfoil section		
Horizonal tail aspect ratio	4.19	4.19
Vertical tail area	8.60	8.80
X-section height incl canopy (ft)	3.25	3.29
X-section height (firewall)	2.50	2.53
X-section width	3.67	3.71
Length overall (ft.)	20.00	20.23
Fuel capacity (usable)	33.00	
Empty weight (lb.)	820.00	848.28
Gross weight	1400.00	1448.28
Useful load	**580.00**	**600.00**
Wheel base (in.)	48.00	48.55
Nose wheel	retractable	
Advertised performance		
Max speed sea level (mph)	225	223.73
75% cruise speed 8k'	210	208.82
Stall speed	55	54.69
ROC sl (ft/min) (GW)	1150	1143.52
Range at 75% (no reserve) nm	1000	
G limits at gross weight	9/-4.5	
Horsepower	118	122.76

TABLE A.2. DRAG POLAR ESTIMATION FOR THE CONVENTIONAL RG

DRAG POLAR BUILDUP

Aircraft **Conventional RG** *Input italicized data*

Wing from NASA tech paper 1865 for the NLF-0215F airfoil
Cdmin = *0.0045* S = *77.7*

Fuselage (ft.)

Length = *20.2* W/L = .183
Firewall height = *2.53* H/L = .125
Max height incl canopy = *3.3*
Max width = *3.7*
Firewall X-section (sq ft) = 9.4
adjust for roundness factor of. *0 9*
Adjusted X-section (sq ft) = 8.4
% for canopy = .30
Adjusted X-section (sq ft) = 11.0

from Smetana Table 5-2, cdπ = *.063*

Horizontal Tail from Smetana Table 5-3, cdπ = *0.0043*

Vertical Tail from Smetana Table 5-3, cdπ = *0.0043*

Component	Cdπ	Aπ	CdπAπ
Wing	.0045	77.7	0.35
Fuselage	.0630	11 0	0.69
Hor. tail	.0043	*10.8*	0.05
Vert. tail	.0043	*8.8*	0 04
Total			1.13

Interference effects *0.06*
Protuberance effects *0 03*

Equiv. flat plate area (sum of CdπAπ and effects) 1.23

CDO = .01579 AR = *7 2*
e = *0.938* K1 = .0158
K3 = .0471

CD = .0158 + .0471 CL^2

TABLE A.3. POINT.TXT OUTPUT FOR THE CONVENTIONAL RG

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.15800D-01 + 0.00000D+00*CL**2 + 0.47100D-01*CL** 0.20000D+01
WING AREA = 0.77740D+02 SQ.FT   WEIGHT = 0.14483D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE
++

MINIMUM LEVEL FLIGHT SPEED = 0.72108D+02 FT/SEC
LIFT COEFFICIENT = 0.30109D+01   DRAG COEFFICIENT = 0.44278D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.31817D+03 FT/SEC
LIFT COEFFICIENT = 0.15465D+00   DRAG COEFFICIENT = 0.16926D-01

MAXIMUM CLIMB ANGLE = 0.65156D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.16442D+03 FT/SEC
LIFT COEFFICIENT = 0.57910D+00   DRAG COEFFICIENT = 0.31595D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.12492D+03 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.11398D+05 FT-LBS/SEC
LIFT COEFFICIENT = 0.10032D+01   DRAG COEFFICIENT = 0.63200D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.16441D+03 FT/SEC
LIFT COEFFICIENT = 0.57919D+00   DRAG COEFFICIENT = 0.31600D-01

SERVICE CEILING = 0.22450D+05 FT
VELOCITY AT SERVICE CEILING = 0.21878D+03 FT/SEC
LIFT COEFFICIENT = 0.66693D+00   DRAG COEFFICIENT = 0.36750D-01

ABSOLUTE CEILING = 0.24622D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.22100D+03 FT/SEC
LIFT COEFFICIENT = 0.70497D+00   DRAG COEFFICIENT = 0.39208D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD
0.00000D+00	0.21396D+02	0.20955D+03	0.49535D+05	0.35651D+00	0.21786D-01
0.50000D+03	0.20884D+02	0.20961D+03	0.48682D+05	0.36158D+00	0.21958D-01
0.10000D+04	0.20376D+02	0.20967D+03	0.47839D+05	0.36672D+00	0.22134D-01

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.74995D-09	0.15358D+05	0.15358D+05	0.72108D+02
0.23087D+01	0.17441D+05	0.14098D+05	0.80000D+02
0.49751D+01	0.20137D+05	0.12932D+05	0.90000D+02
0.74114D+01	0.22875D+05	0.12141D+05	0.10000D+03
0.96526D+01	0.25634D+05	0.11654D+05	0.11000D+03
0.11715D+02	0.28392D+05	0.11425D+05	0.12000D+03
0.13603D+02	0.31127D+05	0.11426D+05	0.13000D+03
0.15315D+02	0.33819D+05	0.11639D+05	0.14000D+03
0.16842D+02	0.36445D+05	0.12053D+05	0.15000D+03
0.18174D+02	0.38983D+05	0.12662D+05	0.16000D+03
0.19298D+02	0.41413D+05	0.13463D+05	0.17000D+03
0.20199D+02	0.43712D+05	0.14457D+05	0.18000D+03
0.20861D+02	0.45859D+05	0.15646D+05	0.19000D+03
0.21266D+02	0.47832D+05	0.17033D+05	0.20000D+03
0.21396D+02	0.49610D+05	0.18622D+05	0.21000D+03
0.21233D+02	0.51170D+05	0.20418D+05	0.22000D+03
0.20759D+02	0.52492D+05	0.22427D+05	0.23000D+03
0.19953D+02	0.53554D+05	0.24656D+05	0.24000D+03
0.18796D+02	0.54333D+05	0.27110D+05	0.25000D+03
0.17269D+02	0.54809D+05	0.29798D+05	0.26000D+03
0.15351D+02	0.54959D+05	0.32725D+05	0.27000D+03
0.13023D+02	0.54762D+05	0.35901D+05	0.28000D+03
0.10264D+02	0.54196D+05	0.39331D+05	0.29000D+03
0.70537D+01	0.53241D+05	0.43025D+05	0.30000D+03
0.33717D+01	0.51873D+05	0.46990D+05	0.31000D+03
0.14659D-09	0.50435D+05	0.50435D+05	0.31817D+03

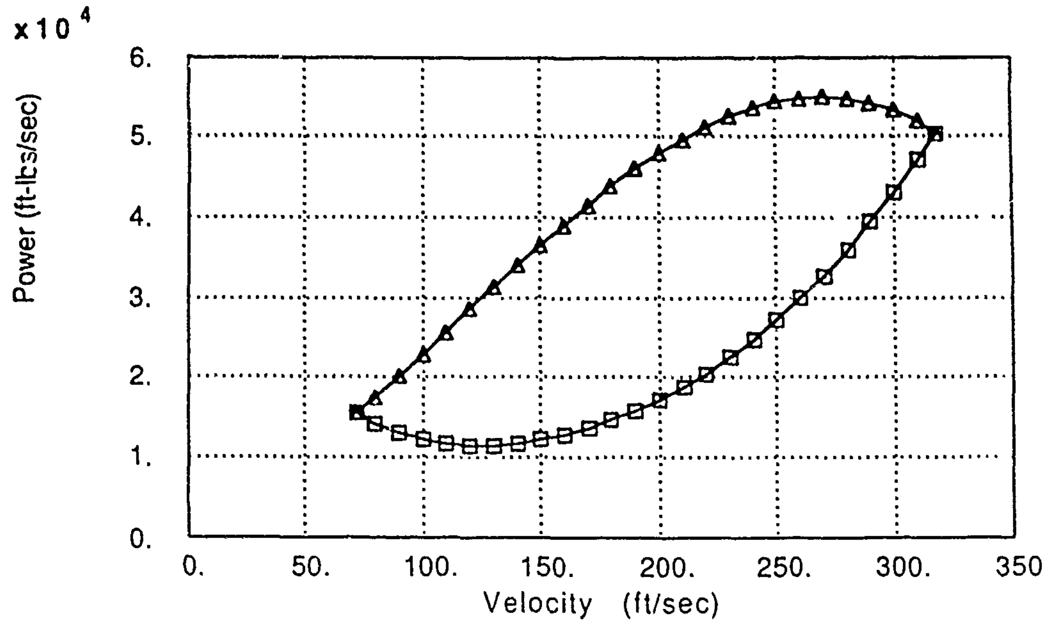


Figure A.1. Power vs Velocity Curve for the Conventional RG

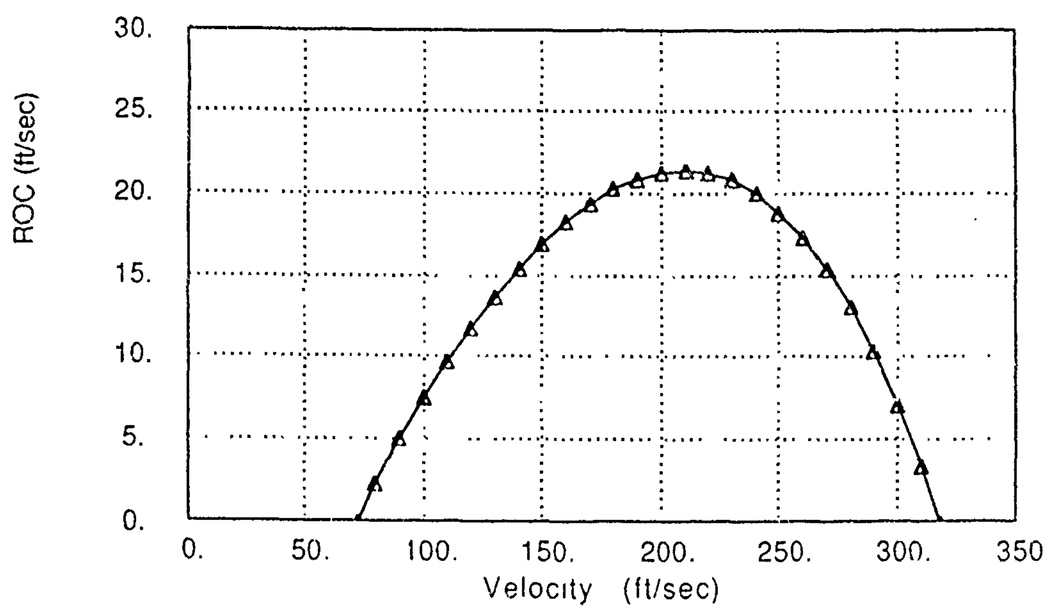


Figure A.2. Rate of Climb vs Velocity Curve for the Conventional RG

APPENDIX B MODEL FOR THE CONVENTIONAL TRACTOR CONFIGURATION WITH FIXED GEAR

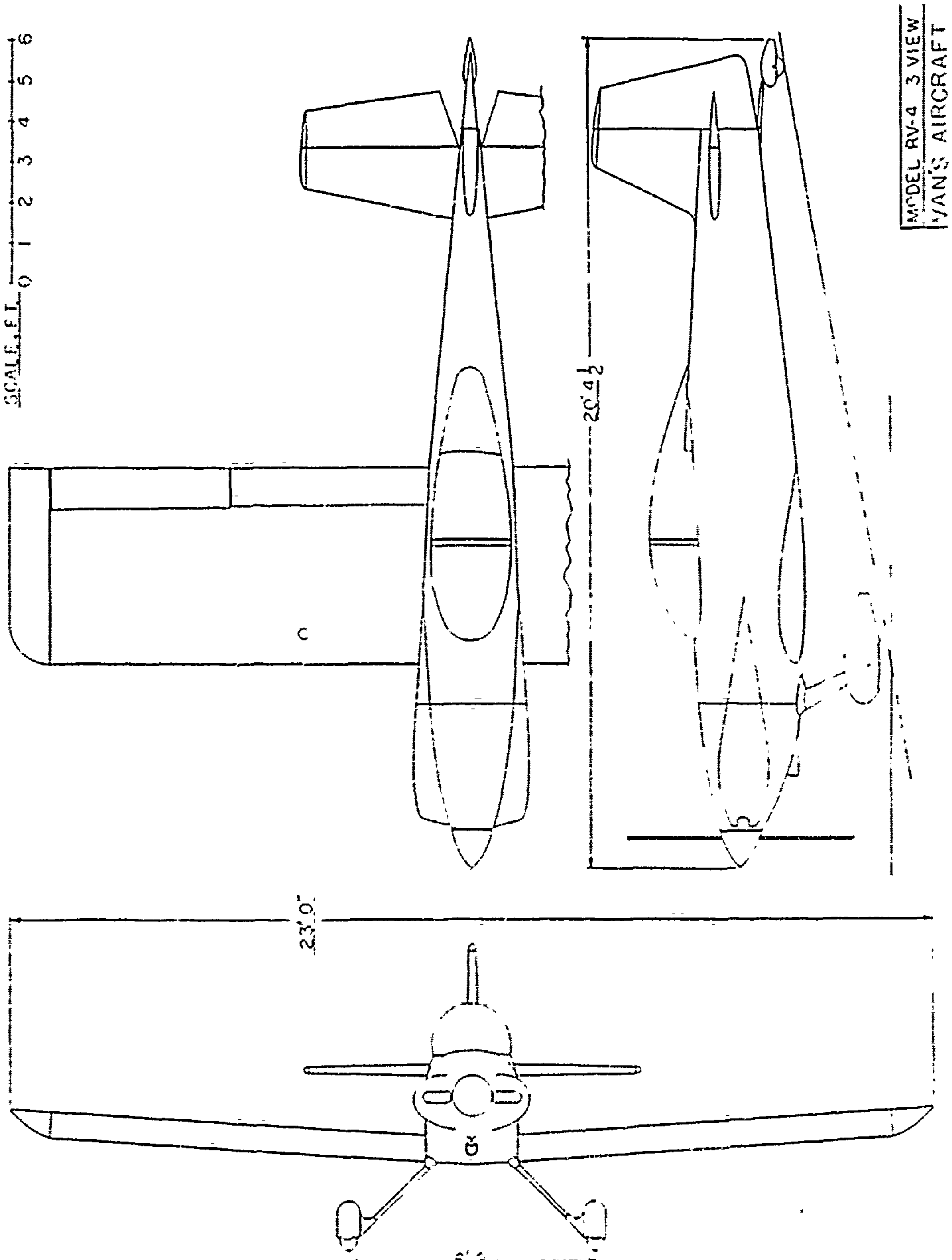


TABLE B.1. SCALED PARAMETERS BASED ON λ FOR THE CONVENTIONAL FG

Conventional fixed gear tractor aircraft based on the RV-4

Aircraft Specifications		λ = **1.01**
	Actual values	Scaled values
Wing span (ft.)	23.00	22.87
Wing chord root	4.90	4.87
Wing chord tip	4.90	4.87
Wing area (sq. ft.)	110.00	108.79
Wing airfoil section	NACA 23013.5	NACA 23013.5
Wing aspect ratio	4.81	4.81
Wing loading (GW) (lb./sq. ft.)	13.64	13.56
Effective horizonal tail span	8.50	8.45
Horizonal tail chord root	3.40	3.38
Horizonal tail chord tip	2.00	1.99
Horizonal tail area (sq. ft.)	21.90	21.66
Horizonal tail airfoil section		
Horizonal tail aspect ratio	3.30	3.30
Vertical tail area	10.87	10.75
X-section height incl canopy	3.70	3.68
X-section height (firewall)	2.60	2.59
X-section width	2.30	2.29
Length overall (ft.)	20.33	20.22
Fuel capacity (usable)	32.00	
Empty weight (lb.)	890.00	875.41
Gross weight	1500.00	1475.41
Useful load	**610.00**	**600.00**
Wheel base (in.)	74.00	73.59
Advertised performance		
Max speed sea level (mph)	201	201.55
75% cruise speed 8k'	186	186.51
Stall speed	54	54.15
ROC sl (ft/min) (GW)	1650	1654.55
Range at 75% (no reserve) nm	650	
G limits at gross weight		
Horsepower	150	147.14

TABLE B.2. DRAG POLAR ESTIMATION FOR THE CONVENTIONAL FG

DRAG POLAR BUILDUP

Aircraft **Conventional FG** *Input italicized data*

Wing from Theory of Wing Sections NACA 23013.5 airfoil
Cdmin = *.0060* S = *108.8*

Fuselage (ft.)

Length = *20.2* W/L = .113
Firewall height = *2.59* H/L = .128
Max height incl canopy = *3.7*
Max width = *2.3*
Firewall X-section (sq ft) = 5.9
adjust for roundness factor of *0.9*
Adjusted X-section (sq ft) = 5.3
% for canopy = .42
Adjusted X-section (sq ft) = 7.6

from Smetana Table 5-2, cdπ = *.071*

Gear from Smetana Table 5-4, CdπAπ = *0.68*

Horizontal Tail from Smetana Table 5-3, cdπ = *0.0043*

Vertical Tail from Smetana Table 5-3, cdπ = *0.0043*

Component	Cdπ	A π	CdπAπ
Wing	.0060	108.8	0.65
Fuselage	.0710	7.6	0.54
Gear			0.68
Engine nacelle	*.1000*	*.8*	0.08
Hor. tail	.0043	*21.9*	0.09
Vert. tail	.0043	*10.9*	0.05
Total			**2.09**

Interference effects *0.06*
Protuberance effects *0.03*

Equiv. flat plate area (sum of CdπAπ and effects) $\boxed{2.28}$

CDO = .0209 AR = *4.81*
e = *0.957* k1 = .0209
k3 = .0692

$\boxed{CD = .0209 + .0692\ CL^2}$

TABLE B.3. POINT.TXT OUTPUT FOR THE CONVENTIONAL FG

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.20900D-01 + 0.00000D+00*CL**2 + 0.69200D-01*CL** 0.20000D+01
WING AREA = 0.10880D+03 SQ.FT   WEIGHT = 0.14754D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE
++

MINIMUM LEVEL FLIGHT SPEED = 0.76279D+02 FT/SEC
LIFT COEFFICIENT = 0.19585D+01   DRAG COEFFICIENT = 0.28633D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.26533D+03 FT/SEC
LIFT COEFFICIENT = 0.16187D+00   DRAG COEFFICIENT = 0.22713D-01

MAXIMUM CLIMB ANGLE = 0.50631D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.14931D+03 FT/SEC
LIFT COEFFICIENT = 0.51114D+00   DRAG COEFFICIENT = 0.38980D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.10941D+03 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.14178D+05 FT-LBS/SEC
LIFT COEFFICIENT = 0.95188D+00   DRAG COEFFICIENT = 0.83600D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.14400D+03 FT/SEC
LIFT COEFFICIENT = 0.54957D+00   DRAG COEFFICIENT = 0.41800D-01

SERVICE CEILING = 0.16587D+05 FT
VELOCITY AT SERVICE CEILING = 0.18565D+03 FT/SEC
LIFT COEFFICIENT = 0.55315D+00   DRAG COEFFICIENT = 0.42074D-01

ABSOLUTE CEILING = 0.18908D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.18745D+03 FT/SEC
LIFT COEFFICIENT = 0.58611D+00   DRAG COEFFICIENT = 0.44672D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD
0.00000D+00	0.14552D+02	0.17908D+03	0.43506D+05	0.35534D+00	0.29638D-01
0.50000D+03	0.14127D+02	0.17917D+03	0.42769D+05	0.36023D+00	0.29880D-01
0.10000D+04	0.13704D+02	0.17926D+03	0.42040D+05	0.36518D+00	0.30128D-01

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.32645D-09	0.16454D+05	0.16454D+05	0.76279D+02
0.10253D+01	0.17441D+05	0.15929D+05	0.80000D+02
0.35496D+01	0.20137D+05	0.14900D+05	0.90000D+02
0.57845D+01	0.22875D+05	0.14341D+05	0.10000D+03
0.77642D+01	0.25634D+05	0.14179D+05	0.11000D+03
0.95028D+01	0.28392D+05	0.14371D+05	0.12000D+03
0.11002D+02	0.31127D+05	0.14895D+05	0.13000D+03
0.12257D+02	0.33819D+05	0.15736D+05	0.14000D+03
0.13255D+02	0.36445D+05	0.16889D+05	0.15000D+03
0.13981D+02	0.38983D+05	0.18355D+05	0.16000D+03
0.14420D+02	0.41413D+05	0.20138D+05	0.17000D+03
0.14550D+02	0.43712D+05	0.22245D+05	0.18000D+03
0.14352D+02	0.45859D+05	0.24684D+05	0.19000D+03
0.13804D+02	0.47832D+05	0.27465D+05	0.20000D+03
0.12884D+02	0.49610D+05	0.30600D+05	0.21000D+03
0.11569D+02	0.51170D+05	0.34102D+05	0.22000D+03
0.98348D+01	0.52492D+05	0.37982D+05	0.23000D+03
0.76580D+01	0.53554D+05	0.42255D+05	0.24000D+03
0.50145D+01	0.54333D+05	0.46935D+05	0.25000D+03
0.18800D+01	0.54809D+05	0.52035D+05	0.26000D+03
0.10396D-09	0.54931D+05	0.54931D+05	0.26533D+03

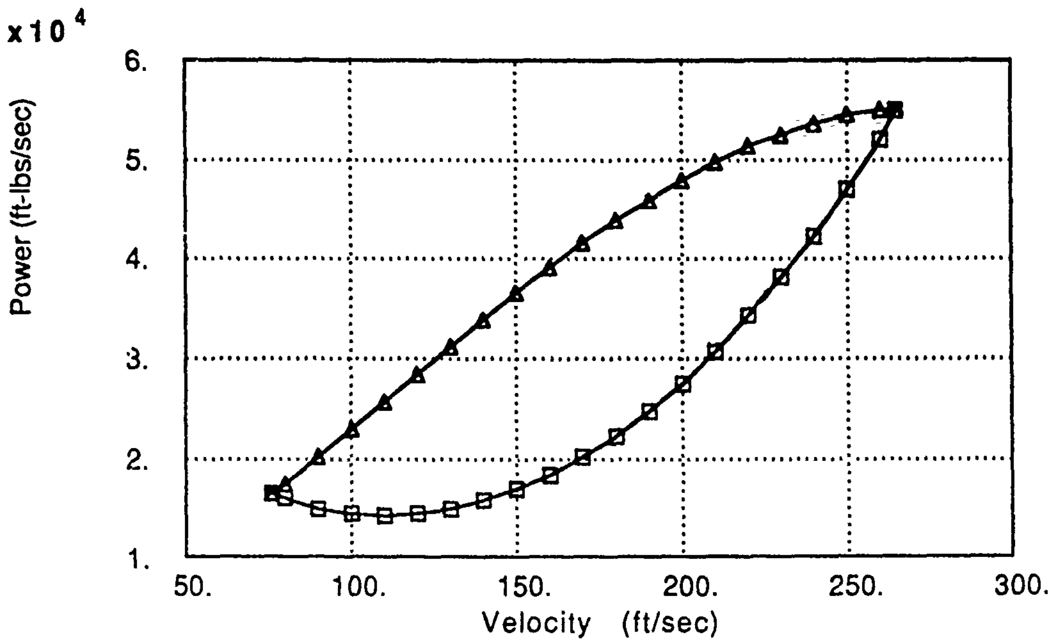


Figure B.1. Power vs Velocity Curve for the Conventional FG

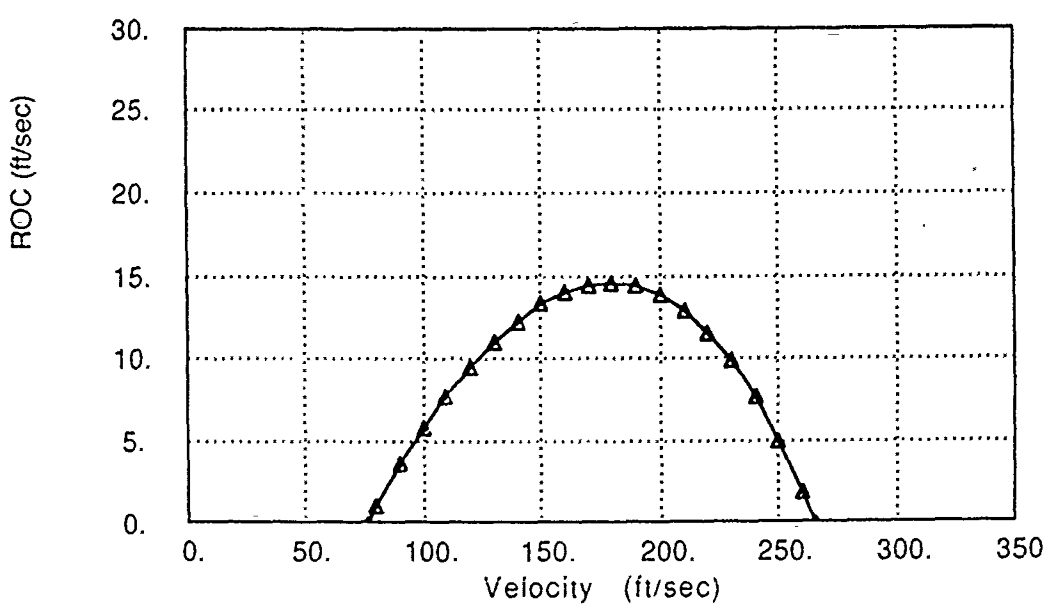


Figure B.2. Rate of Climb vs Velocity Curve for the Conventional FG

APPENDIX C MODEL FOR CONVENTIONAL PUSHER CONFIGURATION WITH RETRACTABLE GEAR

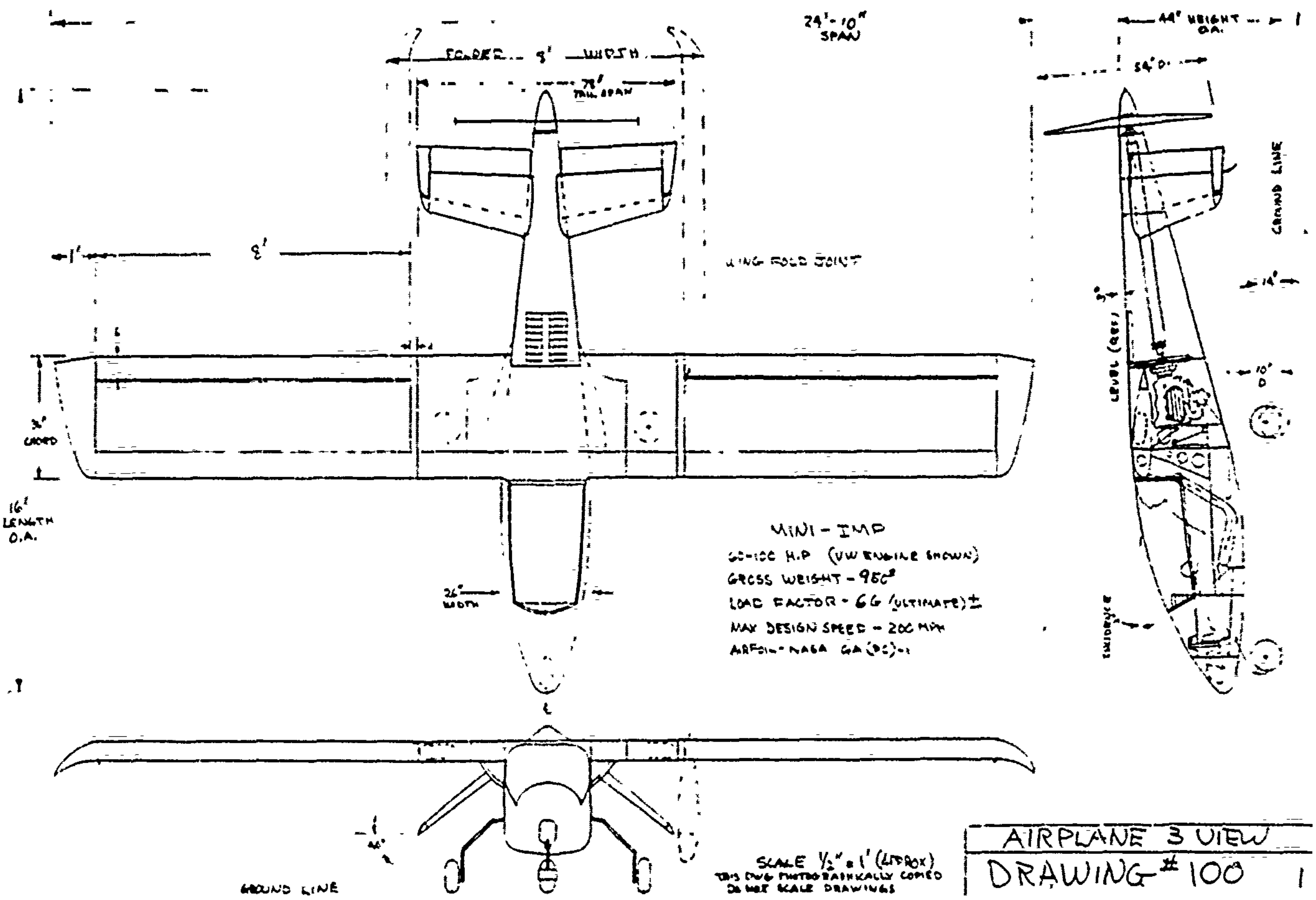


TABLE C.1. SCALED PARAMETERS BASED ON λ FOR THE PUSHER RG

Conventional retractable pusher aircraft based on the Mini-Imp

Aircraft Specifications	Actual values	Scaled values ($\lambda = 0.93$)
Wing span (ft.)	24.83	26.75
Wing chord root	3.00	3.23
Wing chord tip	3.00	3.23
Wing area (sq. ft.)	74.00	85.87
Wing airfoil section	GA (PC) -1	GA (PC) -1
Wing aspect ratio	8.33	8.33
Wing loading (GW) (lb./sq. ft.	13.51	14.56
Effective horizonal tail span	6.50	7.00
Inverted V-tail chord root	2.30	2.48
Inverted V-tail chord tip	1.60	1.72
Inverted V-tail area (sq. ft.)	14.73	17.09
Inverted V-tail airfoil section		
Inverted V-tail aspect ratio	2.87	2.87
X-section height	2.70	2.91
X-section width	2.20	2.37
Length overall (ft.)	14.75	15.89
Fuel capacity (usable)	12.50	
Empty weight (lb.)	520.00	650.00
Gross weight	1000.00	1250.00
Useful load	480.00	600.00
Wheel base (in.)	59.00	63.56
Nose wheel	retractable	
Advertised performance		
Max speed sea level (mph)	200	192.70
75% cruise speed	175	168.61
Stall speed		0.00
ROC sl (ft/min)	1500	1445.24
Range at 75% (no reserve) nm		
G limits at gross weight	4/-3	
Horsepower	100	129.74

TABLE C.2. DRAG POLAR ESTIMATION FOR THE PUSHER RG

DRAG POLAR BUILDUP

Aircraft **Conv. Pusher RG** *Input italicized data*

Wing from McCormick, p. 79 for the GA(PC)-1 airfoil
Cdmin = *0.008* S = *85.9*

Fuselage (ft.)

Length =	*15.9*	W/L = .149
Firewall height =	*2.91*	H/L = .183
Max height incl canopy =	*2.9*	
Max width =	*2.4*	
Firewall X-section (sq ft) =	6.9	
adjust for roundness factor o	*0.95*	
Adjusted X-section (sq ft) =	6.6	
% for canopy =		
Adjusted X-section (sq ft) =	6.6	

from Smetana Table 5-2, cdπ *..063*

Inverted V-Tail from Smetana Table 5-3, cdπ = *0.0043*

Component	Cdπ	Aπ	CdπAπ
Wing	.0080	85.9	0.69
Fuselage	.0630	6.6	0.41
V-tail	.0043	*17.1*	0.07
Total			**1.17**

Interference effects *0.06*
Protuberance effects *0.03*

Equiv flat plate area (sum of CdπAπ and effects) **1.28**

CDO =	.01489	AR =	*8.33*
e =	*0.93*	k1 =	.0149
		k3 =	.0411

CD = .0149 + .0411 CL^2

TABLE C.3. POINT.TXT OUTPUT FOR THE PUSHER RG

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.14900D-01 + 0.00000D+00*CL**2 + 0.41100D-01*CL** 0.20000D+01
WING AREA = 0.85900D+02 SQ.FT WEIGHT = 0.12500D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE
++

MINIMUM LEVEL FLIGHT SPEED = 0.56492D+02 FT/SEC
LIFT COEFFICIENT = 0.38317D+01 DRAG COEFFICIENT = 0.61832D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.31725D+03 FT/SEC
LIFT COEFFICIENT = 0.12150D+00 DRAG COEFFICIENT = 0.15507D-01

MAXIMUM CLIMB ANGLE = 0.82866D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.15111D+03 FT/SEC
LIFT COEFFICIENT = 0.53556D+00 DRAG COEFFICIENT = 0.26688D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.10829D+03 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.77356D+04 FT-LBS/SEC
LIFT COEFFICIENT = 0.10429D+01 DRAG COEFFICIENT = 0.59600D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.14251D+03 FT/SEC
LIFT COEFFICIENT = 0.60210D+00 DRAG COEFFICIENT = 0.29800D-01

SERVICE CEILING = 0.26879D+05 FT
VELOCITY AT SERVICE CEILING = 0.21022D+03 FT/SEC
LIFT COEFFICIENT = 0.65932D+00 DRAG COEFFICIENT = 0.32767D-01

ABSOLUTE CEILING = 0.29124D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.21197D+03 FT/SEC
LIFT COEFFICIENT = 0.70333D+00 DRAG COEFFICIENT = 0.35231D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD
0.00000D+00	0.26049D+02	0.20519D+03	0.48781D+05	0.29044D+00	0.18367D-01
0.50000D+03	0.25486D+02	0.20517D+03	0.47929D+05	0.29477D+00	0.18471D-01
0.10000D+04	0.24929D+02	0.20516D+03	0.47087D+05	0.29918D+00	0.18579D-01

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.49714D-09	0.11395D+05	0.11395D+05	0.56492D+02
0.11715D+01	0.12264D+05	0.10800D+05	0.60000D+02
0.42501D+01	0.14810D+05	0.94972D+04	0.70000D+02
0.70469D+01	0.17441D+05	0.86328D+04	0.80000D+02
0.96370D+01	0.20137D+05	0.80907D+04	0.90000D+02
0.12056D+02	0.22875D+05	0.78054D+04	0.10000D+03
0.14316D+02	0.25634D+05	0.77385D+04	0.11000D+03
0.16420D+02	0.28392D+05	0.78672D+04	0.12000D+03
0.18359D+02	0.31127D+05	0.81788D+04	0.13000D+03
0.20122D+02	0.33819D+05	0.86668D+04	0.14000D+03
0.21693D+02	0.36445D+05	0.93287D+04	0.15000D+03
0.23055D+02	0.38983D+05	0.10165D+05	0.16000D+03
0.24188D+02	0.41413D+05	0.11178D+05	0.17000D+03
0.25071D+02	0.43712D+05	0.12373D+05	0.18000D+03
0.25684D+02	0.45859D+05	0.13753D+05	0.19000D+03
0.26005D+02	0.47832D+05	0.15326D+05	0.20000D+03
0.26010D+02	0.49610D+05	0.17097D+05	0.21000D+03
0.25678D+02	0.51170D+05	0.19074D+05	0.22000D+03
0.24983D+02	0.52492D+05	0.21263D+05	0.23000D+03
0.23905D+02	0.53554D+05	0.23673D+05	0.24000D+03
0.22417D+02	0.54333D+05	0.26311D+05	0.25000D+03
0.20498D+02	0.54809D+05	0.29186D+05	0.26000D+03
0.18122D+02	0.54959D+05	0.32306D+05	0.27000D+03
0.15267D+02	0.54762D+05	0.35679D+05	0.28000D+03
0.11907D+02	0.54196D+05	0.39313D+05	0.29000D+03
0.80184D+01	0.53241D+05	0.43218D+05	0.30000D+03
0.35774D+01	0.51873D+05	0.47401D+05	0.31000D+03
0.18943D-09	0.50612D+05	0.50612D+05	0.31725D+03

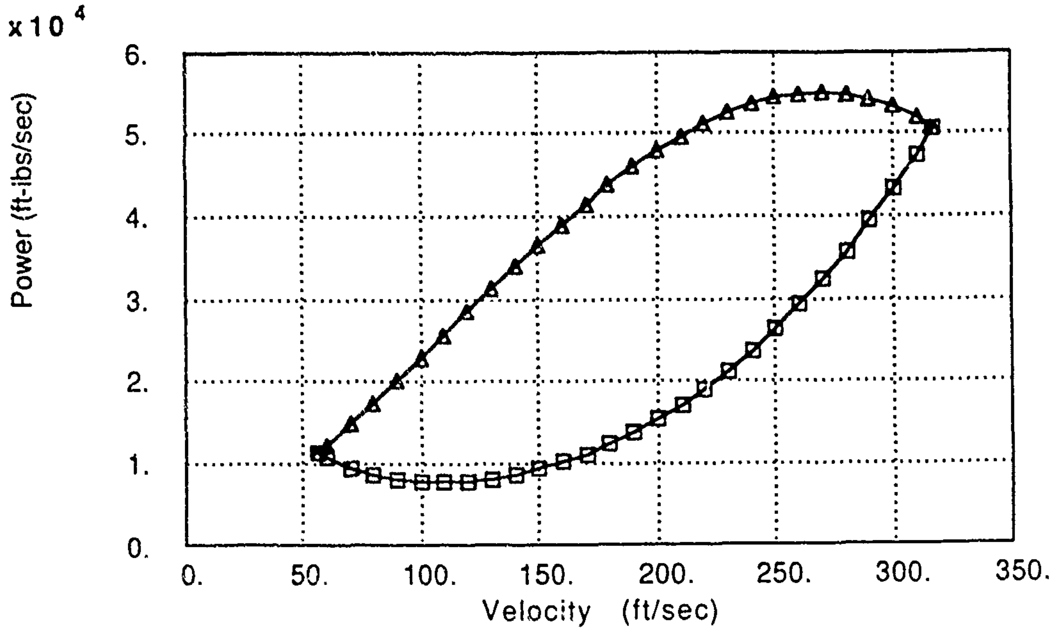


Figure C.1. Power vs Velocity Curve for the Pusher RG

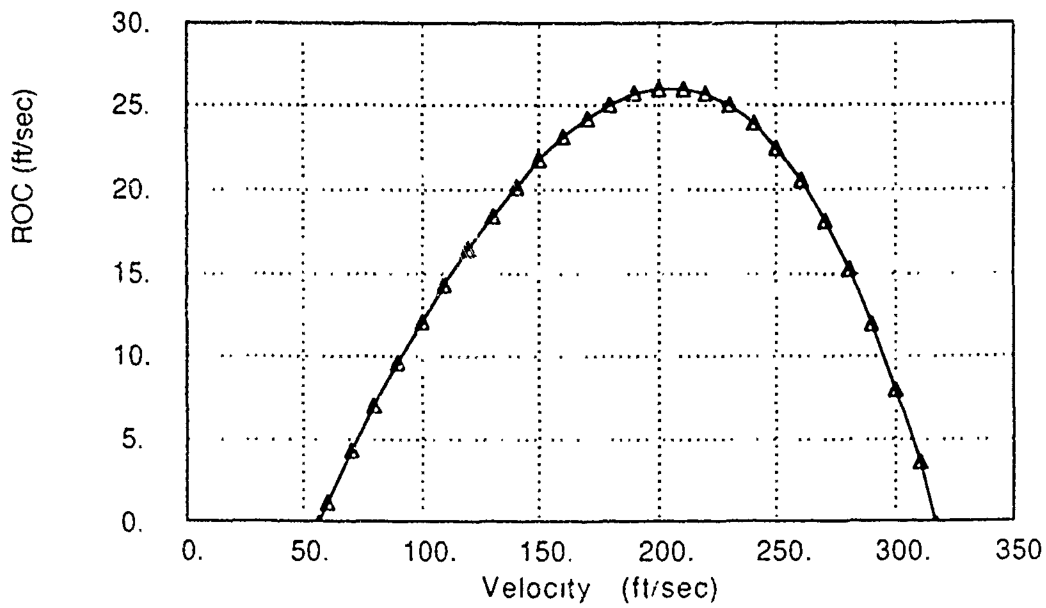


Figure C.2. Rate of Climb vs Velocity Curve for the Pusher RG

APPENDIX D MODEL FOR THE CANARD CONFIGURATION WITH A RETRACTABLE NOSEGEAR

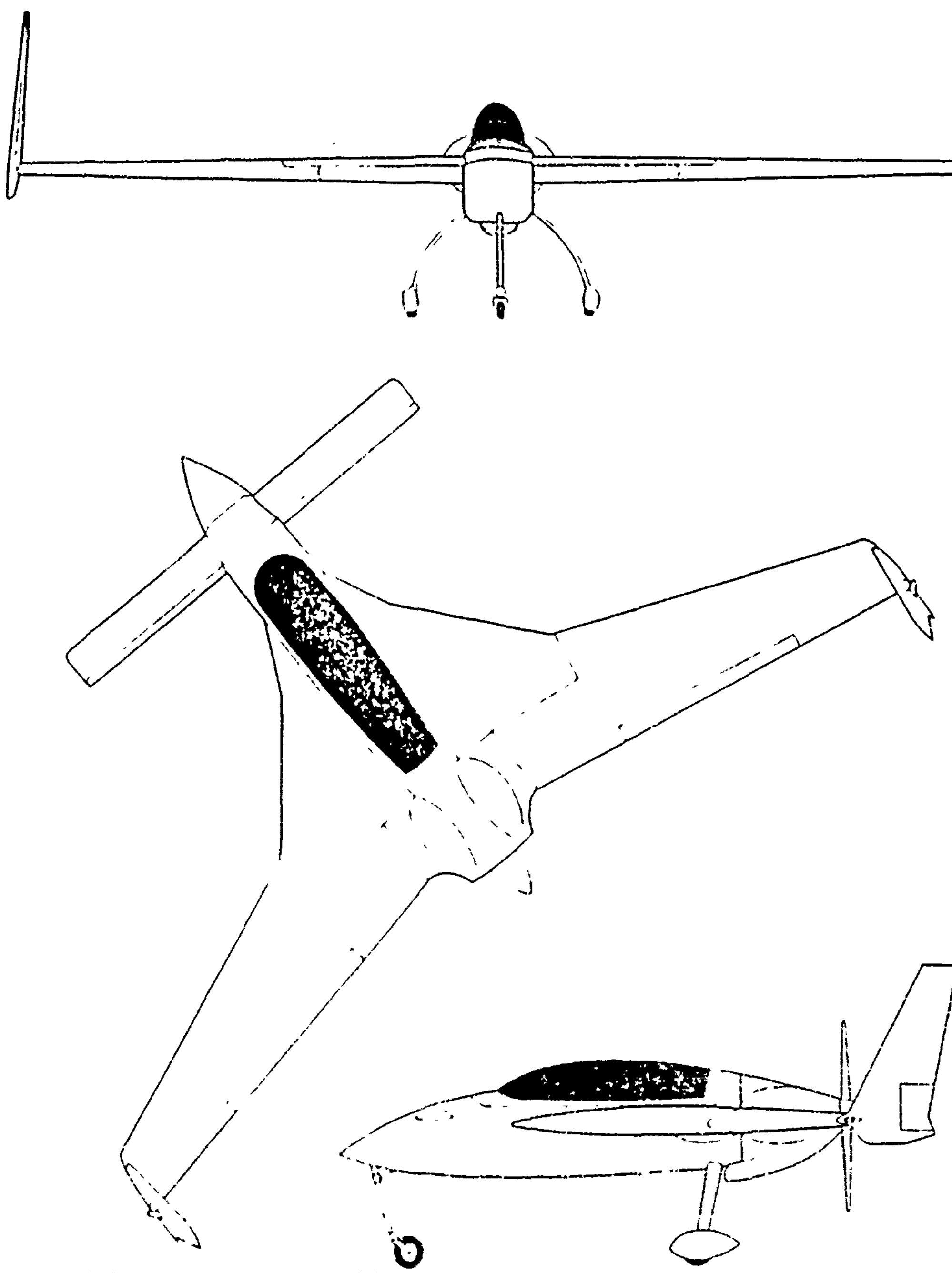


Fig. 3.15 Three view drawing of ... E.

TABLE D.1. SCALED PARAMETERS BASED ON λ FOR THE CANARD

canard aircraft based on the LongEze

Aircraft Specifications	Actual values	Scaled values ($\lambda = 1.01$)
Wing span (ft.)	26.10	25.89
Wing chord root	6.80	6.74
Wing chord tip	1.80	1.79
Main wing area (sq. ft.)	84.24	82.86
Wing airfoil section	Eppler	
Wing aspect ratio	8.09	
Wing loading (GW) (lb./sq. ft.)	7.30	7.24
Canard span (ft.)	11.70	11.60
Canard chord root	1.10	1.09
Canard chord tip	1.10	1.09
Canard area (sq. ft.)	12.87	12.66
Canard airfoil section	GA(W)-1	
Canard aspect ratio	10.64	
Total wing area	97.11	95.52
Vertical winglet area (each)	7.88	7.75
X-section height incl canopy	3.10	3.07
X-section height (firewall)	2.20	2.18
X-section width	1.90	1.88
Length overall (ft.)	16.75	16.61
Fuel capacity (usable)	52.00	
Empty weight (lb.)	710.00	692.68
Gross weight	1325.00	1292.68
Useful load	615.00	600.00
Wheel base (in.)	59.00	58.52
Nose wheel	Retractable	
Advertised performance		
Max speed 8k' (mph)		0.00
75% cruise speed 8k'	183	183.75
Stall speed		0.00
ROC sl (ft/min)	1350	1355.57
Range at 75% (40' reserve) nm	965	
G limits at gross weight		
Horsepower	115	111.73

TABLE D.2. DRAG POLAR ESTIMATION FOR THE CANARD

DRAG POLAR BUILDUP

Aircraft **Canard FG** *Input italicized data*

Wing for Eppler airfoil
Cdmin = *.0080* S = *82.9*
Canard from Eppler for GA(A)-1 airfoil
Cdmin = *.0065* S = *12.7*
Fuselage (ft.) Total S = 95.52
Fuselage length = *12.4* W/L = .152
Firewall height = *2.18* H/L = .176
Max height incl canopy = *3.1*
Max width = *1.9*
Firewall X-section (sq ft) = 4.1
adjust for roundness factor o *0.8*
Adjusted X-section (sq ft) = 3.3
% for canopy = *.41*
Adjusted X-section (sq ft) = 4.6

from Smetana Table 5-2, cdπ : *063*

Gear from Smetana Table 5-4, CdπAπ = *0.68*

Vertical Winglets from Smetana Table 5-3, cdπ = *0.0058*

Component	Cdπ	Aπ	CdπAπ
Wing	.0080	82.9	0.66
Canard	.0065	12.7	0.08
Fuselage	.0630	4.6	0.29
Gear	·		0.68
Engine nacelle & intake	*.1000*	*.9*	0.09
Vert. winglets	.0058	*7.8*	0 05
Total			1.85

Interference effects *0.06*
Protuberance effects *0.03*

Equiv. flat plate area (sum of CdπAπ and effects) 2.02

CD0 = .02113 AR = *8 09*
e = *0.9294* k1 = .0211
k3 = .0423

CD = .0211 + .0423 CL^2

TABLE D.2. DRAG POLAR ESTIMATION FOR THE CANARD

DRAG POLAR BUILDUP

Aircraft **Canard FG** *Input italicized data*

Wing for Eppler airfoil
Cdmin = *.0080* S = *82.9*

Canard from Eppler for GA(A)-1 airfoil
Cdmin = *.0065* S = *12.7*

Fuselage (ft.) Total S = 95.52

Fuselage length = *12.4* W/L = .152
Firewall height = *2.18* H/L = .176
Max height incl canopy = *3.1*
Max width = *1.9*
Firewall X-section (sq ft) = 4.1
adjust for roundness factor o *0.8*
Adjusted X-section (sq ft) = 3.3
% for canopy = .41
Adjusted X-section (sq ft) = 4.6

from Smetana Table 5-2, cdπ : *063*

Gear from Smetana Table 5-4, CdπAπ = *0.68*

Vertical Winglets from Smetana Table 5-3, cdπ = *0.0058*

Component	Cdπ	Aπ	CdπAπ
Wing	.0080	82.9	0.66
Canard	.0065	12.7	0.08
Fuselage	.0630	4.6	0.29
Gear	·		0.68
Engine nacelle & intake	*.1000*	*.9*	0.09
Vert. winglets	.0058	*7.8*	0.05
Total			1.85

Interference effects *0.06*
Protuberance effects *0.03*

Equiv. flat plate area (sum of CdπAπ and effects) $\boxed{2.02}$

CDO = .02113 AR = *8.09*
e = *0.9294* k1 = .0211
k3 = .0423

$\boxed{CD = .0211 + .0423\ CL^2}$

TABLE D.3. POINT.TXT OUTPUT FOR THE CANARD

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.21100D-01 + 0.00000D+00*CL**2 + 0.42300D-01*CL** 0.20000D+01
WING AREA = 0.95520D+02 SQ.FT   WEIGHT = 0.12927D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE

MINIMUM LEVEL FLIGHT SPEED = 0.56600D+02 FT/SEC
LIFT COEFFICIENT = 0.35499D+01   DRAG COEFFICIENT = 0.55416D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.27982D+03 FT/SEC
LIFT COEFFICIENT = 0.14525D+00   DRAG COEFFICIENT = 0.21992D-01

MAXIMUM CLIMB ANGLE = 0.72187D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.13808D+03 FT/SEC
LIFT COEFFICIENT = 0.59644D+00   DRAG COEFFICIENT = 0.36148D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.96419D+02 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.85995D+04 FT-LBS/SEC
LIFT COEFFICIENT = 0.12233D+01   DRAG COEFFICIENT = 0.84400D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.12689D+03 FT/SEC
LIFT COEFFICIENT = 0.70627D+00   DRAG COEFFICIENT = 0.42200D-01

SERVICE CEILING = 0.23588D+05 FT
VELOCITY AT SERVICE CEILING = 0.18556D+03 FT/SEC
LIFT COEFFICIENT = 0.70060D+00   DRAG COEFFICIENT = 0.41862D-01

ABSOLUTE CEILING = 0.26057D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.18704D+03 FT/SEC
LIFT COEFFICIENT = 0.75223D+00   DRAG COEFFICIENT = 0.45036D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD	B
0.00000D+00	0.20328D+02	0.18200D+03	0.44155D+05	0.34332D+00	0.26086D-01	
0.50000D+03	0.19858D+02	0.18198D+03	0.43383D+05	0.34845D+00	0.26236D-01	
0.10000D+04	0.19391D+02	0.18197D+03	0.42619D+05	0.35368D+00	0.26391D-01	

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.40567D-09	0.11422D+05	0.11422D+05	0.56600D+02
0.10687D+01	0.12264D+05	0.10882D+05	0.60000D+02
0.39479D+01	0.14810D+05	0.97064D+04	0.70000D+02
0.65290D+01	0.17441D+05	0.90013D+04	0.80000D+02
0.88798D+01	0.20137D+05	0.86580D+04	0.90000D+02
0.11030D+02	0.22875D+05	0.86170D+04	0.10000D+03
0.12987D+02	0.25634D+05	0.88456D+04	0.11000D+03
0.14748D+02	0.28392D+05	0.93266D+04	0.12000D+03
0 16303D+02	0.31127D+05	0.10053D+05	0.13000D+03
0.17634D+02	0.33819D+05	0.11023D+05	0.14000D+03
0.18724D+02	0.36445D+05	0.12240D+05	0.15000D+03
0.19550D+02	0.38983D+05	0.13711D+05	0.16000D+03
0.20091D+02	0.41413D+05	0.15441D+05	0.17000D+03
0 20322D+02	0.43712D+05	0.17442D+05	0.18000D+03
0.20218D+02	0.45859D+05	0.19724D+05	0.19000D+03
0.19754D+02	0 47832D+05	0.22297D+05	0.20000D+03
0.18904D+02	0.49610D+05	0.25173D+05	0.21000D+03
0 17642D+02	0.51170D+05	0.28365D+05	0.22000D+03
0.15941D+02	0.52492D+05	0.31885D+05	0.23000D+03
0.13775D+02	0.53554D+05	0.35747D+05	0.24000D+03
0.11117D+02	0.54333D+05	0.39963D+05	0.25000D+03
0 79387D+01	54809D+05	0.44546D+05	0.26000D+03
0 42142D+01	0.54959D+05	0.49511D+05	0.27000D+03
0 20094D-11	0.54769D+05	0.54769D+05	0.27982D+03

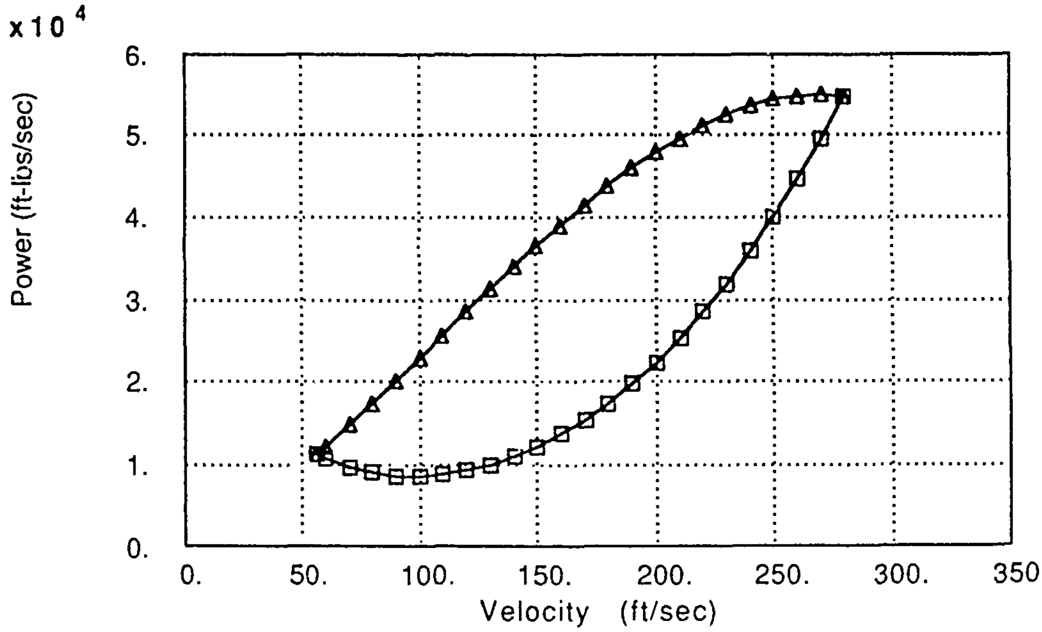


Figure D.1. Power vs Velocity Curve for the Canard

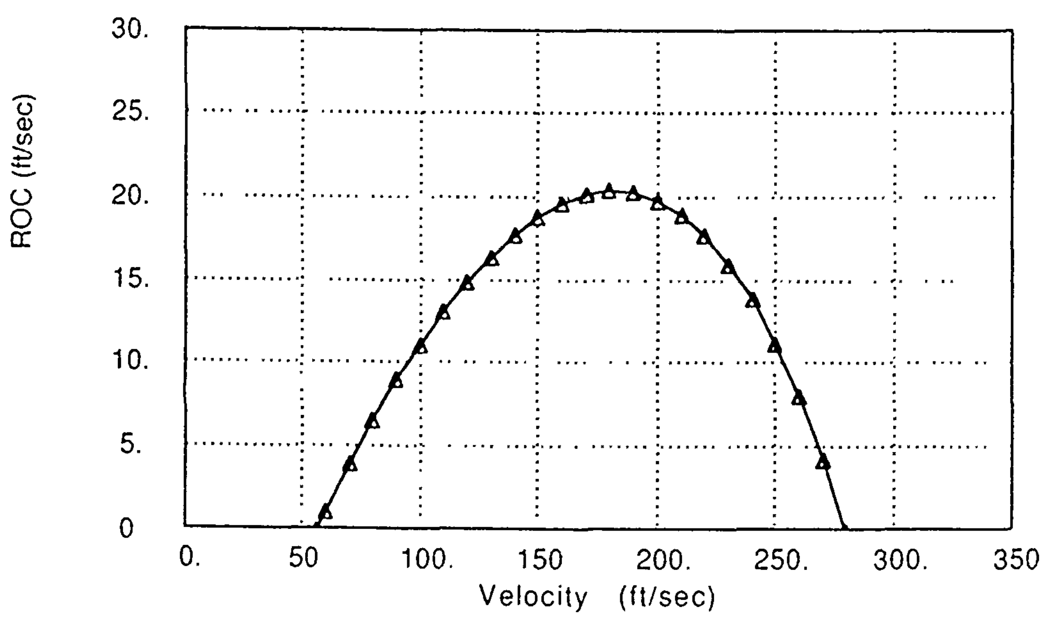


Figure D.2. Rate of Climb vs Velocity Curve for the Canard

APPENDIX E MODEL FOR THE TANDEM-WING CONFIGURATION WITH LOW WING LOADING

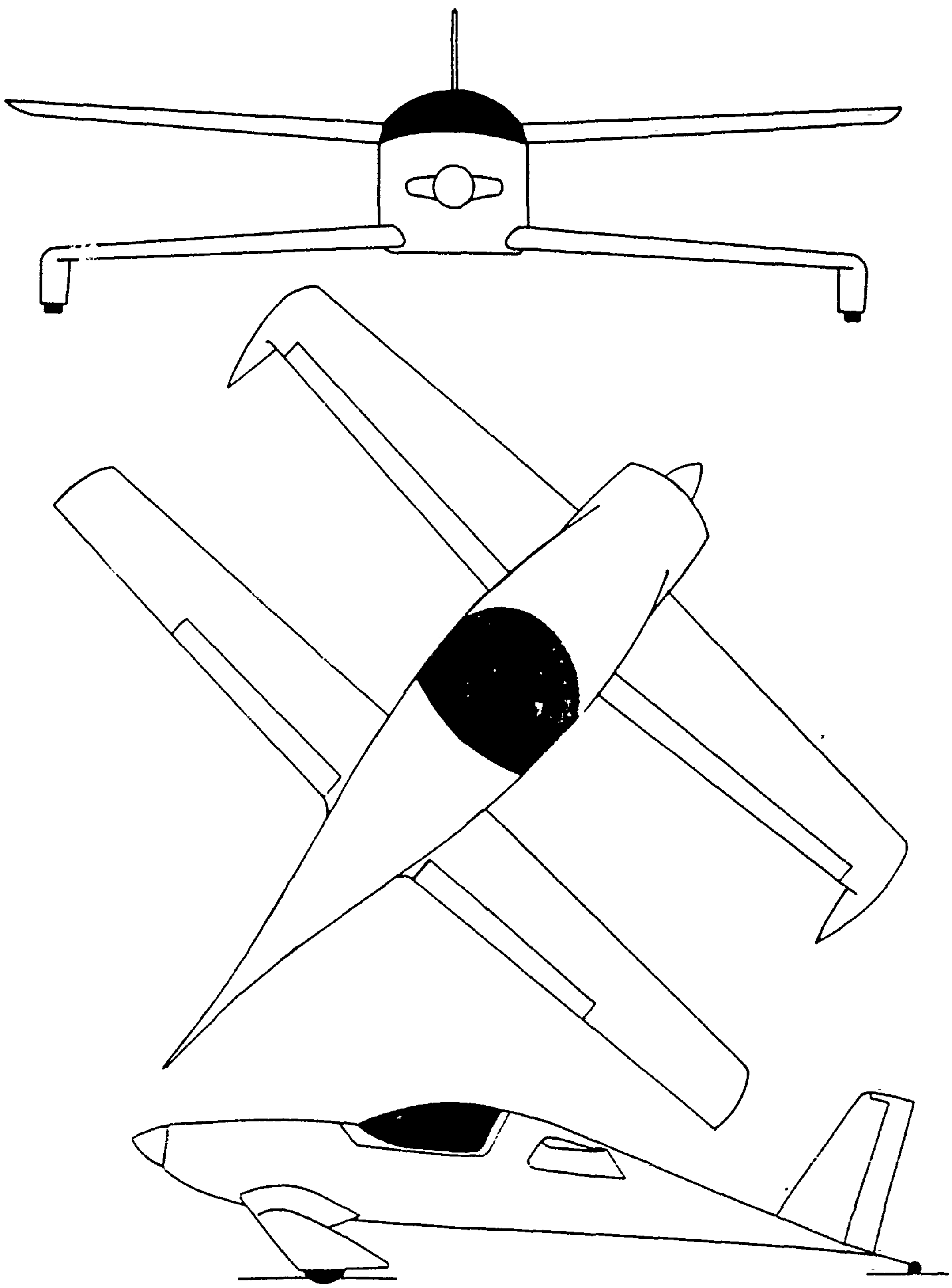


Fig. 3-27 Three view drawing of the Viking Dragonfly

TABLE E.1. SCALED PARAMETERS BASED ON λ FOR THE TANDEM-WING (LOW W/L)

Tandem wing aircraft based on the Dragonfly

Aircraft Specifications	Actual values	Scaled values
		λ = 0.92
Wing span (ft.)	22.00	23.87
Wing chord root	2.60	2.82
Wing chord tip	1.60	1.74
Wing area (sq. ft.)	46.20	54.37
Wing airfoil section (ref)		
Wing aspect ratio	10.48	10.48
Wing loading (GW) (lb./sq. ft.)	11.00	11.93
Canard span (ft.)	20.00	21.70
Canard chord root	3.20	3.47
Canard chord tip	1.70	1.84
Canard area (sq. ft.)	49.00	57.66
Canard airfoil section	GU25-5(11)8 mod	
Canard aspect ratio	8.16	8.16
Total Wing Area	95.20	112.03
Average Aspect ratio	9.29	9.29
X-section height incl canopy (	3.80	4.12
X-section height (firewall)	2.60	2.82
X-section width	3.60	3.91
Vertical tail area	7.12	7.72
Fuselage length (ft.)	18.30	19.85
Fuel capacity (usable)	15.00	
Empty weight (lb.)	605.00	772.34
Gross weight	1075.00	1372.34
Useful load	470.00	600.00
Wheel base (in.)	235.00	254.93
Advertised performance		
Max speed sea level (mph)	168	161.30
75% cruise speed	165	158.42
Stall speed		0.00
ROC sl (ft/min) (GW)	850	816.10
Range at 75% (30' reserve) nm	500	460.91
G limits at gross weight	4.4/-2	
Horsepower	56	74.46

TABLE E.2. DRAG POLAR ESTIMATION FOR THE TANDEM-WING (LOW W/L)

DRAG POLAR BUILDUP

Aircraft **Tandem Wing FG** *Input italicized data*

Wing

Cdmin = *.0090* S = *54.4*

Canard for GU25-5(11)8

Cdmin = *.0090* S = *57.7*

Total S = 112.03

Fuselage (ft.)

Fuselage length =	*19.9*	W/L = .197
Firewall height =	*2.82*	H/L = .142
Max height incl canopy =	*4.1*	
Max width =	*3.9*	
Firewall X-section (sq ft) =	11.0	
adjust for roundness factor of *0.85*		
Adjusted X-section (sq ft) =	9.4	
% for canopy = .46		
Adjusted X-section (sq ft) =	13.7	

from Smetana Table 5-2, $cd\pi$ = *.063*

Gear from Smetana Table 5-4, $Cd\pi A\pi$ = *0.26*

Vertical tail from Smetana Table 5-3 $cd\pi$ = *0.0039*

Component	$Cd\pi$	$A\pi$	$Cd\pi A\pi$
Wing	.0090	54.4	0.49
Canard	.0090	57.7	0.52
Fuselage	.0630	13.7	0.86
Gear			0.26
Vert. tail	.0039	*7.7*	0.03
Total			**2.16**

Interference effects *0.06*

Protuberance effects *0.03*

Equiv flat plate area (sum of $Cd\pi A\pi$ and effects) **2.36**

CDO =	.02292	AR =	*10.48*
e =	*0.91324*	k1 =	.0229
		k3 =	.0333

CD =	.0229	+	.0333	Cl ^2

TABLE E.3. POINT.TXT OUTPUT FOR THE TANDEM-WING (LOW W/L)

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.22900D-01 + 0.00000D+00*CL**2 + 0.33300D-01*CL** 0.20000D+01
WING AREA = 0.11203D+03 SQ.FT   WEIGHT = 0.13723D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE

MINIMUM LEVEL FLIGHT SPEED = 0.49912D+02 FT/SEC
LIFT COEFFICIENT = 0.41321D+01   DRAG COEFFICIENT = 0.59146D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.25886D+03 FT/SEC
LIFT COEFFICIENT = 0.15363D+00   DRAG COEFFICIENT = 0.23686D-01

MAXIMUM CLIMB ANGLE = 0.66927D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.12511D+03 FT/SEC
LIFT COEFFICIENT = 0.65763D+00   DRAG COEFFICIENT = 0.37302D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.84657D+02 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.74091D+04 FT-LBS/SEC
LIFT COEFFICIENT = 0.14363D+01   DRAG COEFFICIENT = 0.91600D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.11141D+03 FT/SEC
LIFT COEFFICIENT = 0.82927D+00   DRAG COEFFICIENT = 0.45800D-01

SERVICE CEILING = 0.23553D+05 FT
VELOCITY AT SERVICE CEILING = 0.16856D+03 FT/SEC
LIFT COEFFICIENT = 0.76758D+00   DRAG COEFFICIENT = 0.42519D-01

ABSOLUTE CEILING = 0.26539D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.16980D+03 FT/SEC
LIFT COEFFICIENT = 0.84050D+00   DRAG COEFFICIENT = 0.46424D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD	B
0.00000D+00	0.17242D+02	0.16746D+03	0.40806D+05	0.36710D+00	0.27388D-01	
0.50000D+03	0.16846D+02	0.16741D+03	0.40086D+05	0.37271D+00	0.27526D-01	
0.10000D+04	0.16454D+02	0.16737D+03	0.39374D+05	0.37842D+00	0.27669D-01	

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.99101D-09	0.98046D+04	0.98046D+04	0.49912D+02
0.25816D-01	0.98255D+04	0.97900D+04	0.50000D+02
0.27429D+01	0.12264D+05	0.84998D+04	0.60000D+02
0.51316D+01	0.14810D+05	0.77674D+04	0.70000D+02
0.72854D+01	0.17441D+05	0.74434D+04	0.80000D+02
0.92430D+01	0.20137D+05	0.74525D+04	0.90000D+02
0.11016D+02	0.22875D+05	0.77571D+04	0.10000D+03
0.12602D+02	0.25634D+05	0.83400D+04	0.11000D+03
0.13988D+02	0.28392D+05	0.91956D+04	0.12000D+03
0.15158D+02	0.31127D+05	0.10326D+05	0.13000D+03
0.16090D+02	0.33819D+05	0.11737D+05	0.14000D+03
0.16763D+02	0.36445D+05	0.13440D+05	0.15000D+03
0.17152D+02	0.38983D+05	0.15445D+05	0.16000D+03
0.17231D+02	0.41413D+05	0.17766D+05	0.17000D+03
0.16974D+02	0.43712D+05	0.20418D+05	0.18000D+03
0.16354D+02	0.45859D+05	0.23416D+05	0.19000D+03
0.15344D+02	0.47832D+05	0.26776D+05	0.20000D+03
0.13915D+02	0.49610D+05	0.30513D+05	0.21000D+03
0.12041D+02	0.51170D+05	0.34646D+05	0.22000D+03
0.96929D+01	0.52492D+05	0.39190D+05	0.23000D+03
0.68422D+01	0.53554D+05	0.44164D+05	0.24000D+03
0.34607D+01	0.54333D+05	0.49584D+05	0.25000D+03
0.63670D-10	0.54770D+05	0.54770D+05	0.25886D+03

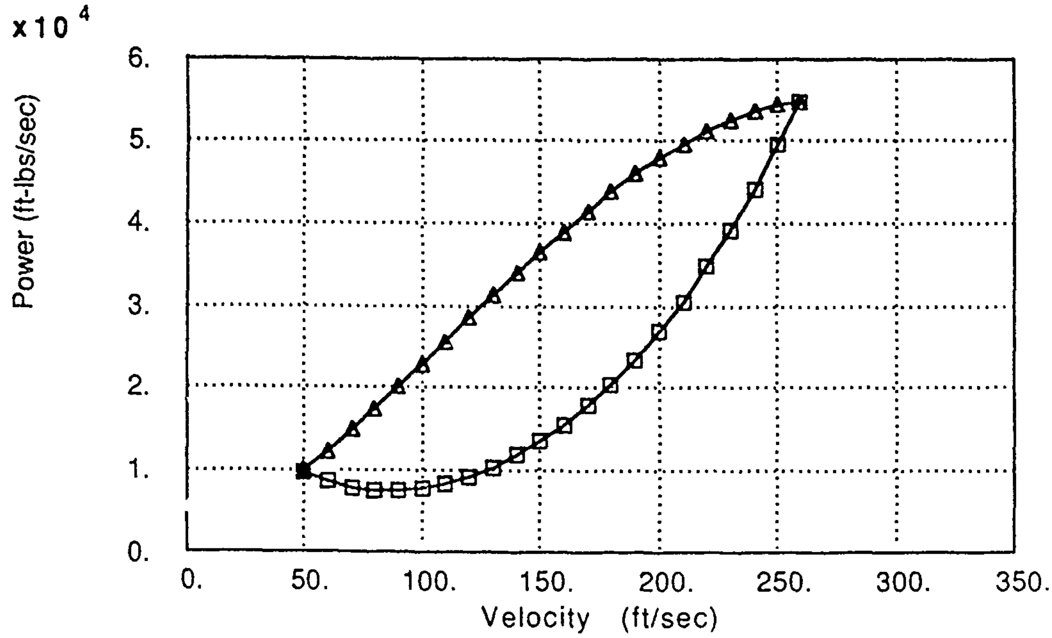


Figure E.1. Power vs Velocity Curve for the Tandem-wing with Low W/L

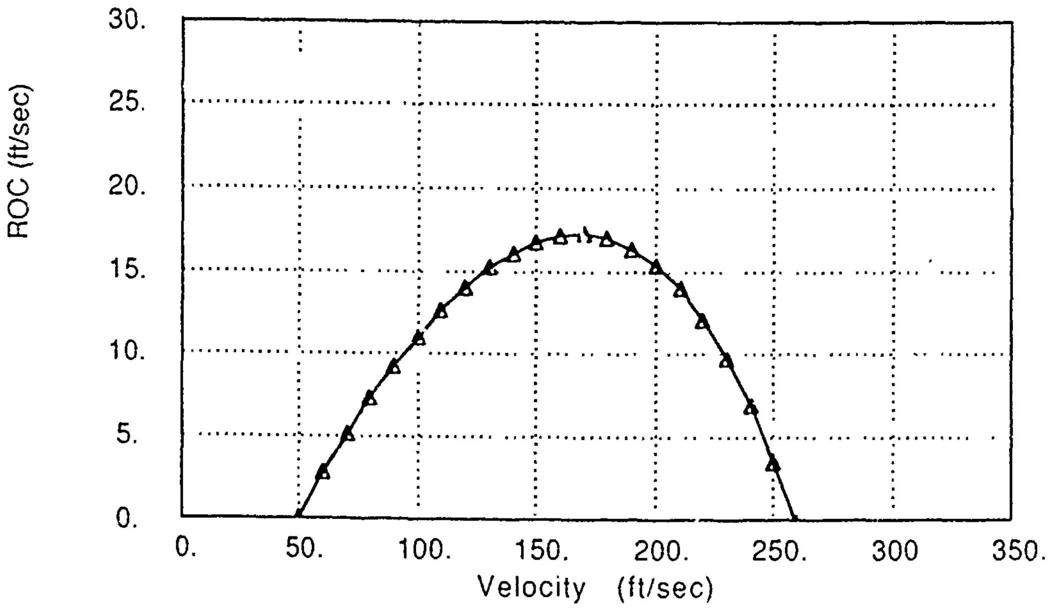


Figure E.2. Rate of Climb vs Velocity Curve for the Tandem-wing with Low W/L

APPENDIX F MODEL FOR THE TANDEM-WING CONFIGURATION WITH HIGH WING LOADING

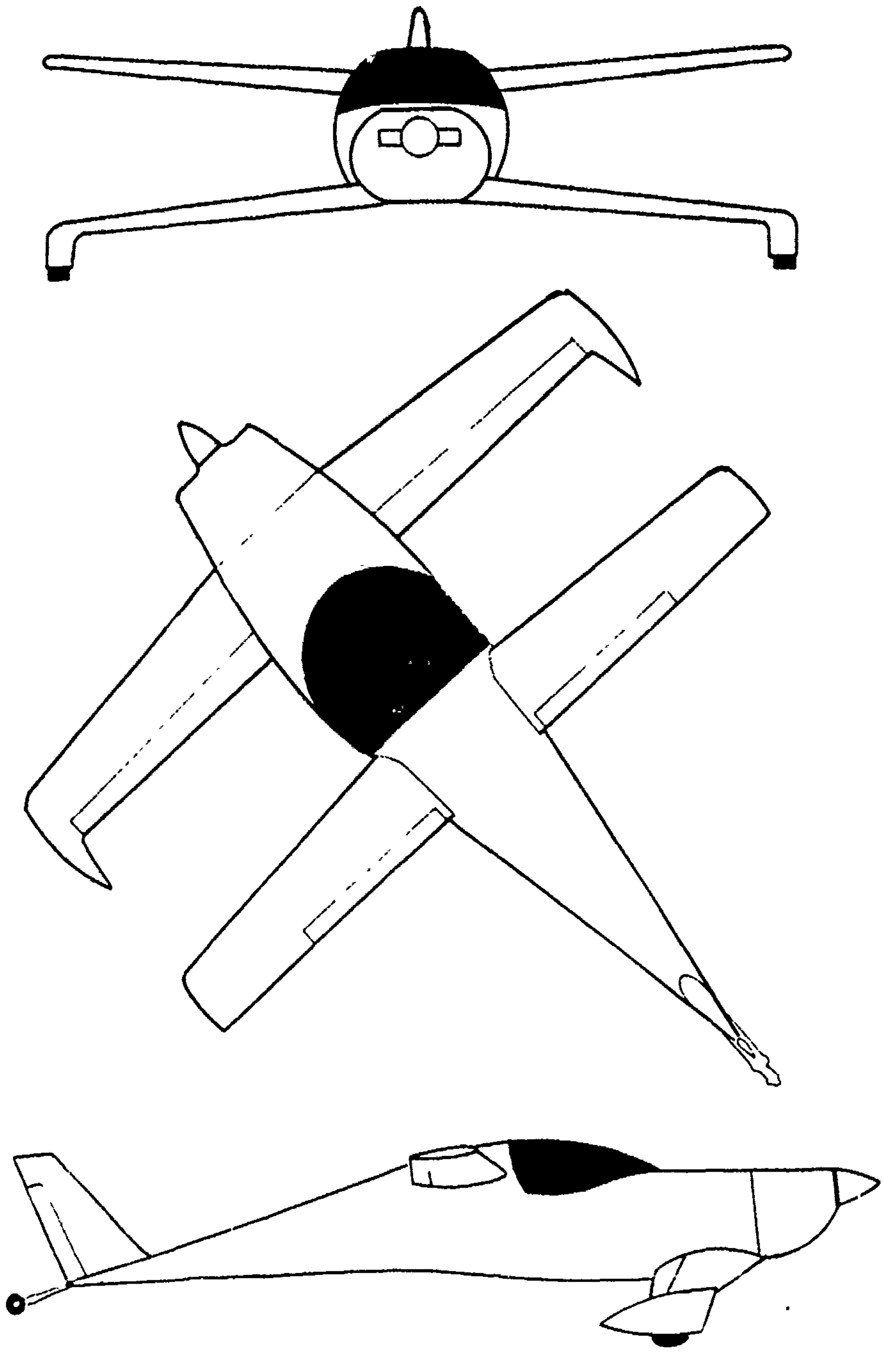


Fig 3-24 Three view drawing of the Quickie Q 200

TABLE F.1. SCALED PARAMETERS BASED ON λ FOR THE TANDEM-WING (HIGH W/L)

Tandem wing aircraft based on the Q-200

Aircraft Specifications		λ = 1.00
	Actual values	Scaled values
Wing span (ft.)	16.67	16.72
Wing chord root	2.20	2.21
Wing chord tip	1.50	1.50
Wing area (sq. ft.)	32.00	32.18
Wing airfoil section (ref)	NASA LS(1)0417MOD	
Wing aspect ratio	8.68	8.68
Wing loading (GW) (lb./sq. ft.)	16.42	16.47
Canard span (ft.)	16.67	16.72
Canard chord root	2.50	2.51
Canard chord tip	1.50	1.50
Canard area (sq. ft.)	35.00	35.20
Canard airfoil section	NASA LS(1)0417MOD	
Canard aspect ratio	7.94	7.94
Total Wing Area	67.00	67.37
Average Aspect ratio	8.30	8.30
X-section height incl canopy (	3.30	3.31
X-section height (firewall)	2.20	2.21
X-section width	3.90	3.91
Vertical tail area	4.16	4.17
Fuselage length (ft.)	18.00	18.05
Fuel capacity (usable)	20.00	
Empty weight (lb.)	505.00	509.24
Gross weight	1100.00	1109.24
Useful load	595.00	600.00
Wheel base (in.)		0.00
Advertised performance		
Max speed sea level (mph)	220	219.69
75% cruise speed	207	206.71
Stall speed	64	63.91
ROC sl (ft/min) (GW)	1200	1198.33
Range at 75% (30' reserve) nm		0.00
G limits at gross weight		
Horsepower	100	100.98

TABLE F.2. DRAG POLAR ESTIMATION FOR THE TANDEM-WING (HIGH W/L)

DRAG POLAR BUILDUP

Aircraft Tandem Wing FG (Q-200) *Input italicized data*

Wing for NASA LS(1)0417mod
Cdmin = *.0078* S = *32.2*

Canard for NASA LS(1)0417mod
Cdmin = *.0078* S = *35.2*

Total S = 67.38

Fuselage (ft.)

Fuselage length = *18.0* W/L = .217
Firewall height = *2.21* H/L = .123
Max height incl canopy = *3.3*
Max width = *3.9*
Firewall X-section (sq ft) = 8.6
adjust for roundness factor of *0.8*
Adjusted X-section (sq ft) = 6.9
% for canopy = .50
Adjusted X-section (sq ft) = 10.4

from Nicholei Figure 8.1, cdπ = *.050*

Gear from Smetana Table 5-4, CdπAπ = *0.26*

Vertical tail from Smetana Table 5-3, cdπ = *0.0039*

Component	Cdπ	A π	Cdπ Aπ
Wing	.0078	32.2	0.25
Canard	.0078	35.2	0.27
Fuselage	.0500	10.4	0.52
Gear			0.26
Vert. tail	.0039	*4.2*	0.02
Total			1.32

Interference effects *0.06*
Protuberance effects *0.03*

Equiv. flat plate area (sum of CdπAπ and effects) $\boxed{1.44}$

CDO = .02327 AR = *8.68*
e = *0.924* k1 = .0233
k3 = .0397

$\boxed{CD = .0233 + .0397}$

TABLE F.3. POINT.TXT OUTPUT FOR THE TANDEM-WING (HIGH W/L)

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.23300D-01 + 0.00000D+00*CL**2 + 0.39700D-01*CL** 0.20000D+01
WING AREA = 0.67380D+02 SQ.FT   WEIGHT = 0.11092D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE
++

MINIMUM LEVEL FLIGHT SPEED = 0.55840D+02 FT/SEC
LIFT COEFFICIENT = 0.44367D+01   DRAG COEFFICIENT = 0.80477D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.30124D+03 FT/SEC
LIFT COEFFICIENT = 0.15245D+00   DRAG COEFFICIENT = 0.24223D-01

MAXIMUM CLIMB ANGLE = 0.89928D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.14490D+03 FT/SEC
LIFT COEFFICIENT = 0.65892D+00   DRAG COEFFICIENT = 0.40537D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.10211D+03 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.79552D+04 FT-LBS/SEC
LIFT COEFFICIENT = 0.13269D+01   DRAG COEFFICIENT = 0.93200D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.13438D+03 FT/SEC
LIFT COEFFICIENT = 0.76610D+00   DRAG COEFFICIENT = 0.46600D-01

SERVICE CEILING = 0.26001D+05 FT
VELOCITY AT SERVICE CEILING = 0.19918D+03 FT/SEC
LIFT COEFFICIENT = 0.80524D+00   DRAG COEFFICIENT = 0.49042D-01

ABSOLUTE CEILING = 0.28064D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.20059D+03 FT/SEC
LIFT COEFFICIENT = 0.85498D+00   DRAG COEFFICIENT = 0.52320D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD	B
0.00000D+00	0.26948D+02	0.19507D+03	0.46883D+05	0.36354D+00	0.28547D-01	
0.50000D+03	0.26353D+02	0.19505D+03	0.46062D+05	0.36899D+00	0.28705D-01	
0.10000D+04	0.25763D+02	0.19503D+03	0.45252D+05	0.37454D+00	0.28869D-01	

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.51402D-09	0.11235D+05	0.11235D+05	0.55840D+02
0.15389D+01	0.12264D+05	0.10557D+05	0.60000D+02
0.49278D+01	0.14810D+05	0.93437D+04	0.70000D+02
0.79962D+01	0.17441D+05	0.85716D+04	0.80000D+02
0.10824D+02	0.20137D+05	0.81309D+04	0.90000D+02
0.13446D+02	0.22875D+05	0.79603D+04	0.10000D+03
0.15875D+02	0.25634D+05	0.80249D+04	0.11000D+03
0.18109D+02	0.28392D+05	0.83050D+04	0.12000D+03
0.20137D+02	0.31127D+05	0.87907D+04	0.13000D+03
0.21944D+02	0.33819D+05	0.94779D+04	0.14000D+03
0.23510D+02	0.36445D+05	0.10367D+05	0.15000D+03
0.24813D+02	0.38983D+05	0.11460D+05	0.16000D+03
0.25829D+02	0.41413D+05	0.12762D+05	0.17000D+03
0.26533D+02	0.43712D+05	0.14280D+05	0.18000D+03
0.26900D+02	0.45859D+05	0.16021D+05	0.19000D+03
0.26901D+02	0.47832D+05	0.17992D+05	0.20000D+03
0.26511D+02	0.49610D+05	0.20203D+05	0.21000D+03
0.25701D+02	0.51170D+05	0.22662D+05	0.22000D+03
0.24442D+02	0.52492D+05	0.25380D+05	0.23000D+03
0.22708D+02	0.53554D+05	0.28365D+05	0.24000D+03
0.20469D+02	0.54333D+05	0.31628D+05	0.25000D+03
0.17696D+02	0.54809D+05	0.35179D+05	0.26000D+03
0.14361D+02	0.54959D+05	0.39029D+05	0.27000D+03
0.10435D+02	0.54762D+05	0.43187D+05	0.28000D+03
0.58880D+01	0.54196D+05	0.47665D+05	0.29000D+03
0.69177D+00	0.53241D+05	0.52473D+05	0.30000D+03
0.24400D-09	0.53094D+05	0.53094D+05	0.30124D+03

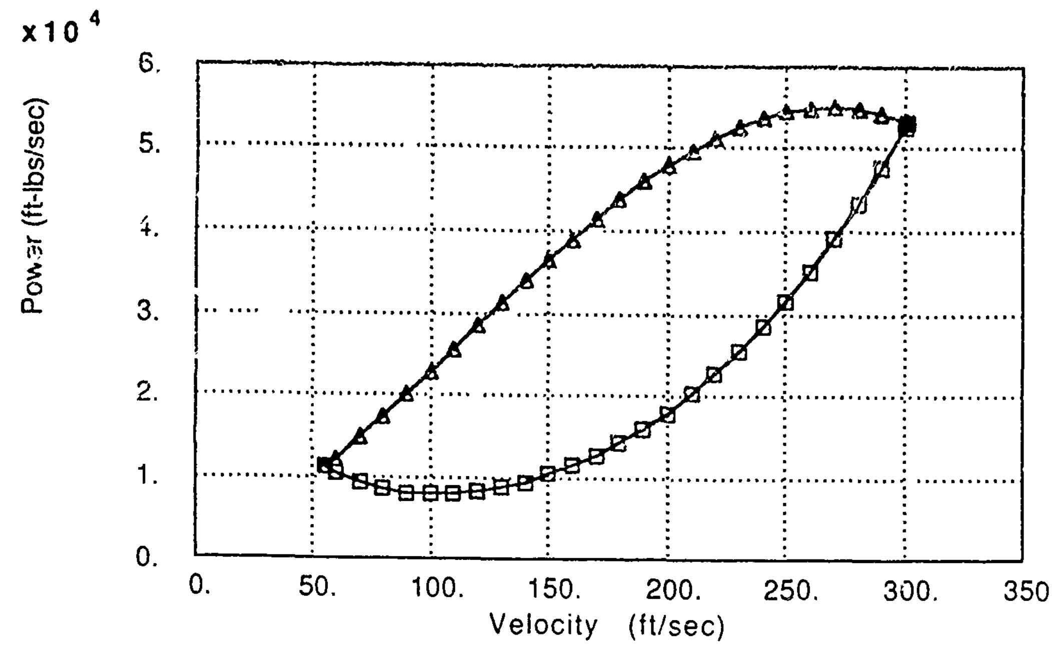


Figure F.1. Power vs Velocity Curve for the Tandem-wing with High W/L

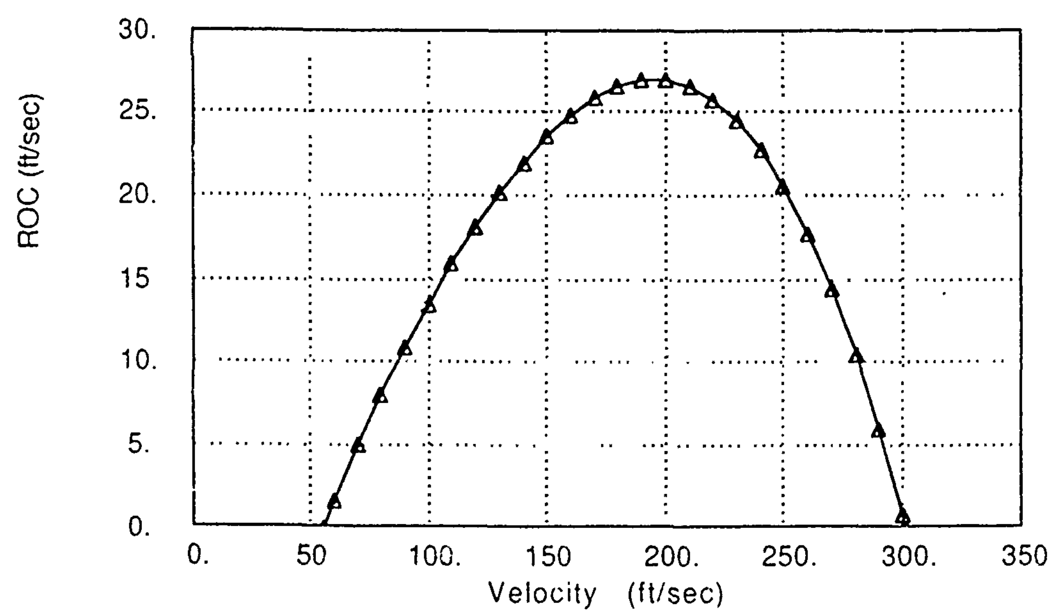


Figure F.2. Rate of Climb vs Velocity Curve for the Tandem-wing with Low W/L

APPENDIX G MODEL FOR THE JOINED-WING CONFIGURATION WITH BICYCLE GEAR

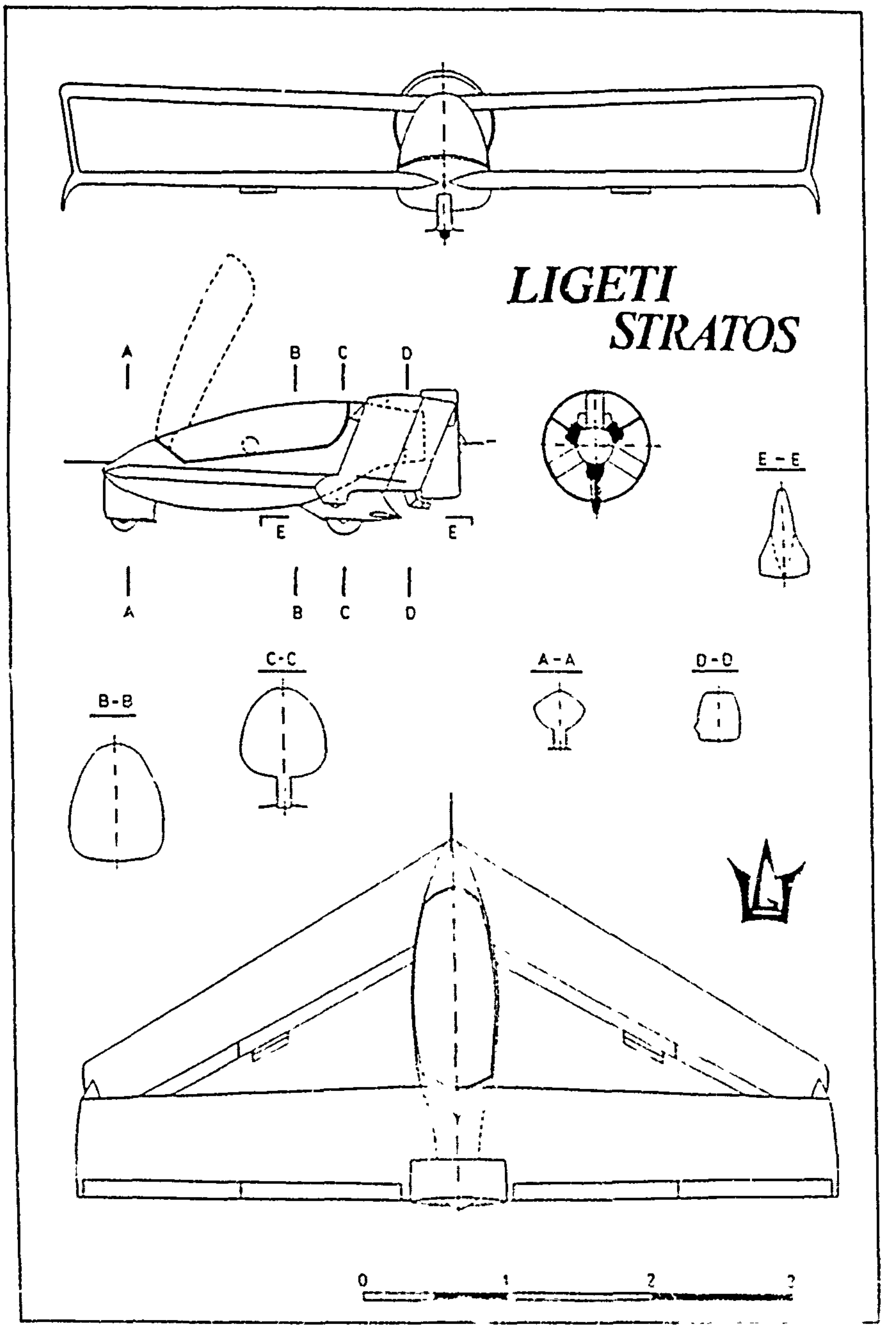


TABLE G.1. SCALED PARAMETERS BASED ON λ FOR THE JOINED-WING CONFIGURATION

Joined wing aircraft based on the Ligeti Stratos

Aircraft Specifications		$\lambda = 0.74$
	Actual values	Scaled values
Wing span (ft.)	17.58	23.79
Wing chord root	2.54	3.44
Wing chord tip	2.13	2.88
Wing area (sq. ft.)	41.00	75.11
Wing airfoil section	Wortmann	
Wing aspect ratio	7.54	7.54
Wing loading (GW) (lb./sq. ft.)	5.13	6.94
Canard span (ft.)	17.00	23.01
Canard chord root	2.54	3.44
Canard chord tip	2.13	2.88
Canard area (sq. ft.)	40.00	73.27
Canard airfoil section	Wortmann	
Card aspect ratio	7.23	7.23
Total wing area	80.70	148.38
Vertical winglet area (each)	6.00	8.12
X-section height incl canopy	2.70	3.65
X-section width	2.00	2.71
Length overall (ft.)	8.17	11.06
Fuel capacity (usable)	5.00	
Empty weight (lb.)	172.00	426.45
Gross weight	414.00	1026.45
Useful load	242.00	600.00
Wheel base (in.)	0.00	0.00
Bicycle gear with tip wheels		
Advertised performance		
Max speed sea level (mph)	124	106.59
75% cruise speed	112	96.27
Stall speed	38	32.66
ROC sl (ft/min)	670	575.91
Range at 75% (no reserve) nm	259	
Vne	167	143.55
G limits at gross weight	9/-6	
Horsepower	28	80.76

TABLE G.2. DRAG POLAR ESTIMATION FOR THE JOINED-WING CONFIGURATION

DRAG POLAR BUILDUP

Aircraft **Joined Wing FG** *Input italicized data*

Wing for FX 66 series Wortmann airfoils
Cdmin = *.0065* S = *75.1*

Canard for FX 66 series Wortmann airfoils
Cdmin = *.0065* S = *73.3*

Total S = 148.37

Fuselage (ft.)

Fuselage length = *11 1* W/L = .245
Firewall height = *3.65* H/L = .330
Max height incl canopy = *3.7*
Max width = *2.7*
Firewall X-section (sq ft) = 9.9
adjust for roundness factor of *0.8*
Adjusted X-section (sq ft) = 7 9
% for canopy =
Adjusted X-section (sq ft) = 7.9

from Nicolai, p 8-6 cdπ = *.042*

Gear from Smetana Table 5-4, CdπAπ = *0.29*

Vertical tails from Smetana Table 5-3, cdπ = *0.0039*

Component	Cdπ	Aπ	CdπAπ
Wing	.0065	75.1	0.49
Canard	.0065	73.3	0.48
Fuselage	.0420	7.9	0.33
Gear			0.29
Two vert. tails	.0039	*16.2*	0.06
Total			**1.65**

Interference effects *0 06*
Protuberance effects *0.03*

Equiv flat plate area (sum of CdπAπ and effects) $\boxed{1.80}$

CDO = .01321 AR = *7.54*
e = *0.938* k1 = .0132
k3 = 0450

$\boxed{CD = .0132 + 0450 \; CL^2}$

TABLE G.3. POINT.TXT OUTPUT FOR THE JOINED-WING CONFIGURATION

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.13200D-01 + 0.00000D+00*CL**2 + 0.45000D-01*CL** 0.20000D-01
WING AREA = 0.14840D+03 SQ.FT   WEIGHT = 0.10265D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE

+++

MINIMUM LEVEL FLIGHT SPEED = 0.38308D+02 FT/SEC
LIFT COEFFICIENT = 0.39608D+01   DRAG COEFFICIENT = 0.71914D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.28439D+03 FT/SEC
LIFT COEFFICIENT = 0.71867D-01   DRAG COEFFICIENT = 0.13432D-01

MAXIMUM CLIMB ANGLE = 0.10300D+02 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.12373D+03 FT/SEC
LIFT COEFFICIENT = 0.37964D+00   DRAG COEFFICIENT = 0.19686D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.78715D+02 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.45477D+04 FT-LBS/SEC
LIFT COEFFICIENT = 0.93808D+00   DRAG COEFFICIENT = 0.52800D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.10359D+03 FT/SEC
LIFT COEFFICIENT = 0.54160D+00   DRAG COEFFICIENT = 0.26400D-01

SERVICE CEILING = 0.30306D+05 FT
VELOCITY AT SERVICE CEILING = 0.17560D+03 FT/SEC
LIFT COEFFICIENT = 0.50875D+00   DRAG COEFFICIENT = 0.24847D-01

ABSOLUTE CEILING = 0.32861D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.17586D+03 FT/SEC
LIFT COEFFICIENT = 0.55796D+00   DRAG COEFFICIENT = 0.27210D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD	B
0.00000D+00	0.27890D+02	0.18105D+03	0.43946D+05	0.17732D+00	0.14615D-01	
0.50000D+03	0.27320D+02	0.18096D+03	0.43160D+05	0.18012D+00	0.14660D-01	
0.10000D+04	0.26755D+02	0.18086D+03	0.42383D+05	0.18298D+00	0.14707D-01	

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.17727D-08	0.71394D+04	0.71394D+04	0.38308D+02
0.63792D+00	0.75159D+04	0.68611D+04	0.40000D+02
0.40572D+01	0.98255D+04	0.56609D+04	0.50000D+02
0.70981D+01	0.12264D+05	0.49781D+04	0.60000D+02
0.99127D+01	0.14810D+05	0.46349D+04	0.70000D+02
0.12560D+02	0.17441D+05	0.45495D+04	0.80000D+02
0.15056D+02	0.20137D+05	0.46824D+04	0.90000D+02
0.17399D+02	0.22875D+05	0.50158D+04	0.10000D+03
0.19573D+02	0.25634D+05	0.55434D+04	0.11000D+03
0.21556D+02	0.28392D+05	0.62654D+04	0.12000D+03
0.23324D+02	0.31127D+05	0.71866D+04	0.13000D+03
0.24848D+02	0.33819D+05	0.83141D+04	0.14000D+03
0.26097D+02	0.36445D+05	0.96572D+04	0.15000D+03
0.27042D+02	0.38983D+05	0.11226D+05	0.16000D+03
0.27650D+02	0.41413D+05	0.13032D+05	0.17000D+03
0.27888D+02	0.43712D+05	0.15086D+05	0.18000D+03
0.27724D+02	0.45859D+05	0.17402D+05	0.19000D+03
0.27124D+02	0.47832D+05	0.19991D+05	0.20000D+03
0.26054D+02	0.49610D+05	0.22866D+05	0.21000D+03
0.24481D+02	0.51170D+05	0.26042D+05	0.22000D+03
0.22371D+02	0.52492D+05	0.29529D+05	0.23000D+03
0.19690D+02	0.53554D+05	0.33343D+05	0.24000D+03
0.16402D+02	0.54333D+05	0.37497D+05	0.25000D+03
0.12475D+02	0.54809D+05	0.42003D+05	0.26000D+03
0.78738D+01	0.54959D+05	0.46877D+05	0.27000D+03
0.25638D+01	0.54762D+05	0.52130D+05	0.28000D+03
0.19226D-09	0.54560D+05	0.54560D+05	0.28439D+03

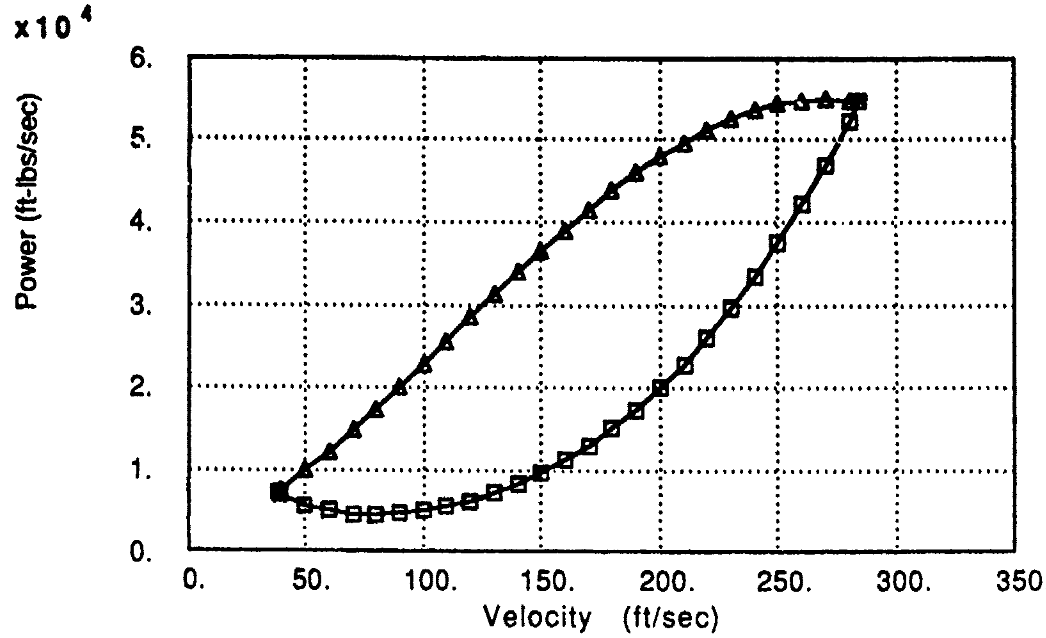


Figure G.1. Power vs Velocity Curve for the Joined-wing

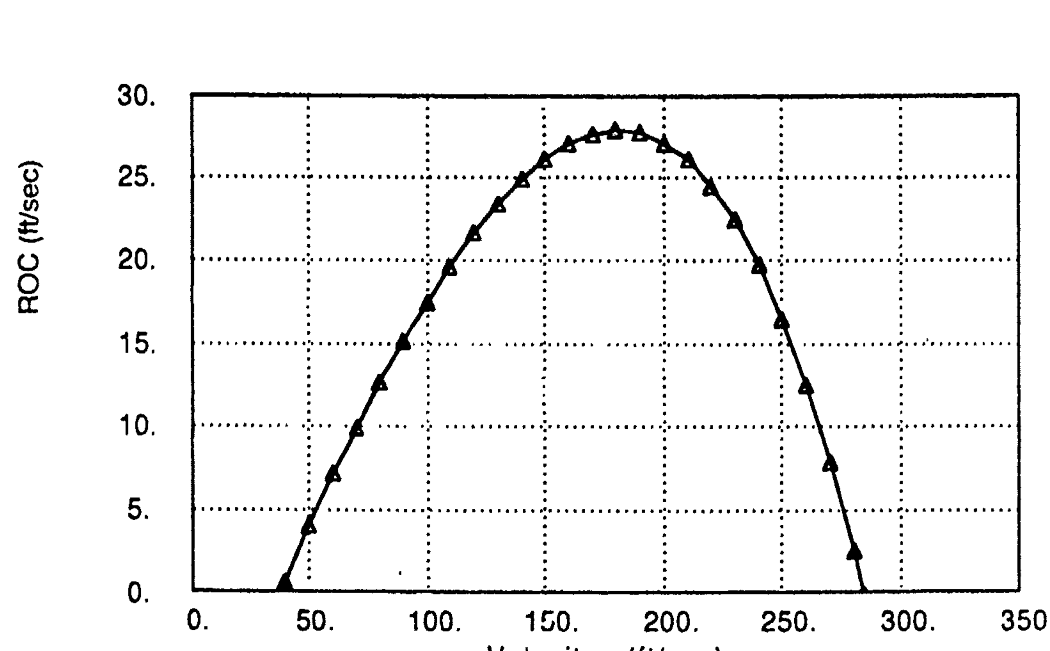
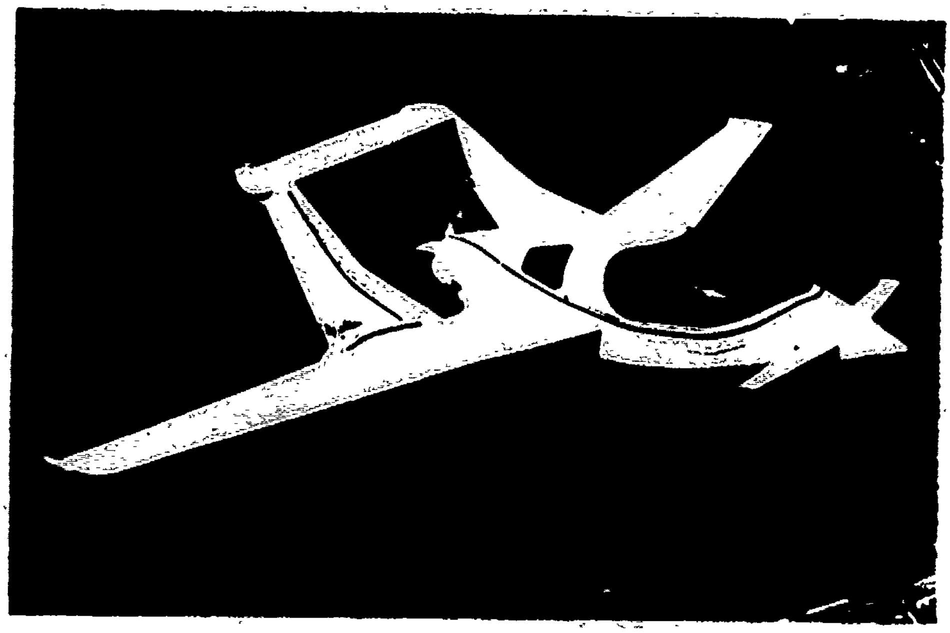


Figure G.2. Rate of Climb vs Velocity Curve for the Joined-wing

APPENDIX H MODEL FOR THE 3-SURFACE CONFIGURATION WITH RETRACTABLE NOSEGEAR



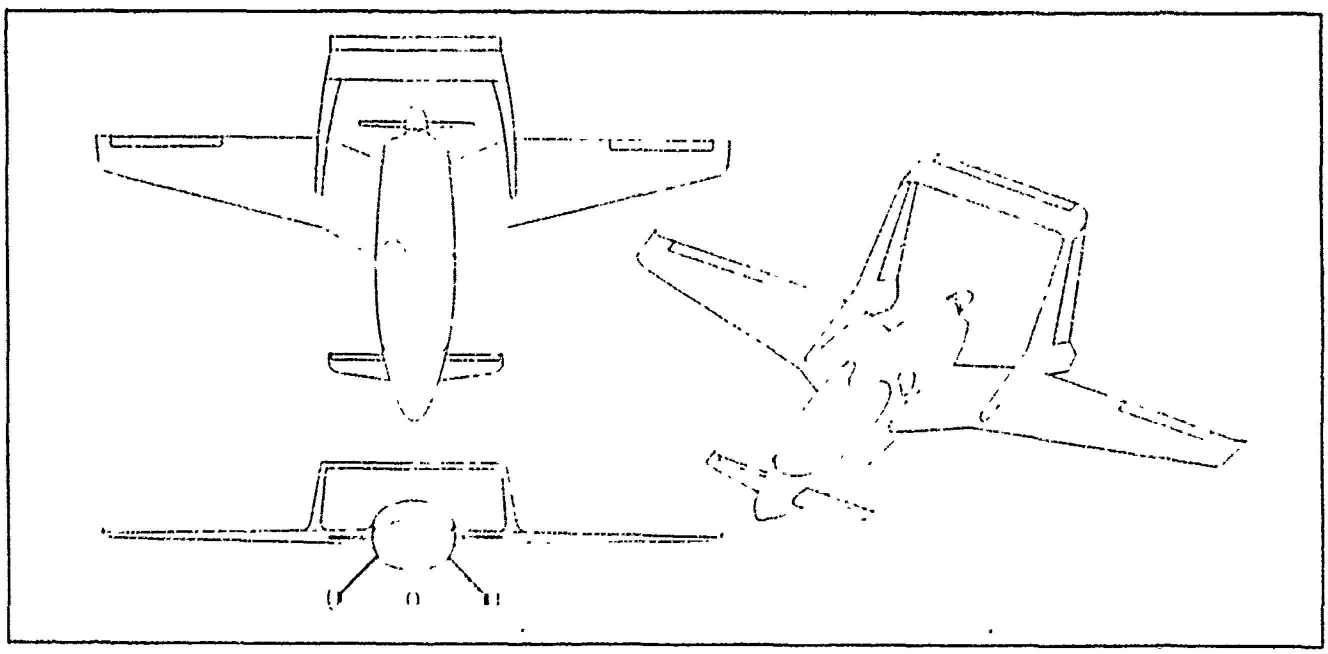


TABLE H.1. SCALED PARAMETERS BASED ON λ FOR THE 3-SURFACE CONFIGURATION

Aircraft Specifications		$\lambda = 1.02$
	Actual values	Scaled values
Wing span (ft.)	30.00	29.36
Wing chord root	4.91	4.81
Wing chord tip	1.67	1.63
Wing area (sq. ft.)	102.50	98.18
Wing airfoil section	NASA (NLF-0215)	
Wing aspect ratio	8.78	8.78
Wing loading (GW) (lb./sq. ft.	16.00	15.66
Canard span (ft.)	7.90	7.73
Canard chord root	1.33	1.30
Canard chord tip	0.67	0.66
Canard area (sq. ft.)	7.90	7.57
Canard airfoil section	NASA (NLF-0215)	
Canard aspect ratio	8.35	8.35
Total wing area	110.40	105.75
Horizonal tail span (ft.)	7.90	7.73
Horizonal tail chord	2.00	1.96
Horizonal tail area (sq. ft.)	15.80	15.13
Horizonal tail airfoil section	NACA (63218)	
Horizonal tail aspect ratio	3.96	3.96
Vertical tail area (each)	5.17	5.06
X-section height incl canopy	3.21	3.14
X-section height (firewall)	3.21	3.14
X-section width	3.50	3.43
Length overall (ft.)	17.92	17.54
Fuselage length	14.50	14.19
Fuel capacity (usable)	30.00	
Empty weight (lb.)	1040.00	975.00
Gross weight	1680.00	1575.00
Useful load	640.00	600.00
Wheel base (in.)	61.00	59.70
Nose wheel	Retractable	
Advertised performance		
Max speed sea level (mph)	190	192.05
75% cruise speed	170	171.84
Stall speed	70	70.76
ROC sl (ft/min)	1500	1516.22
Range at 75% (no reserve) nm	1000	
Va	150	151.62
Vne	250	252.70
Vgear ext	165	166.78
G limits at gross weight	4.4/-2.2	
Horsepower	115	106.66

TABLE H.2. DRAG POLAR ESTIMATION FOR THE 3-SURFACE CONFIGURATION

DRAG POLAR BUILDUP

Aircraft **3-Surface FG** *Input italicized data*

Wing for NLF-0215 airfoil
Cdmin = *.0045* S = *98.2*

Canard for NLF-0215 airfoil
Cdmin = *.0045* S = *7.6*
Total S = 105.75

Fuselage (ft.)

Fuselage length =	*14.2*	W/L = .242
Firewall height =	*3.14*	H/L = .221
Max height incl canopy =	*3 1*	
Max width =	*3.4*	
Firewall X-section (sq ft) =	10.8	
adjust for roundness factor of.	*0.8*	
Adjusted X-section (sq ft) =	8.6	
% for canopy =		
Adjusted X-section (sq ft) =	8.6	

from Nicolai p 8-6, $cd\pi$ = *.050*

Gear from Smetana Table 5-4. $Cd\pi A\pi$ = *0 74*

Vertical tails from Smetana Table 5-3. $cd\pi$ = *0.0058*

Component	$Cd\pi$	$A\pi$	$Cd\pi A\pi$
Wing	.0045	98.2	0 44
Canard	.0045	7.6	0 03
Fuselage	.0500	8 6	0 43
Gear			0.74
Vert. winglets	.0058	*7.8*	0.05
Total			1.69

Interference effects	*0 06*
Protuberance effects	*0.03*

Equiv flat plate area (sum of $Cd\pi A\pi$ and effects) **1.84**

CDO =	.01744	AR =	*8 78*
e =	*0 923*	k1 =	.0174
		k3 =	.0393

CD = 0174 + 0393 CL^2

TABLE H.3. POINT.TXT OUTPUT FOR THE 3-SURFACE CONFIGURATION

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D+05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.18600D-01 + 0.00000D+00*CL**2 + 0.39300D-01*CL** 0.20000D+01
WING AREA = 0.10575D+03 SQ.FT   WEIGHT = 0.15750D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUDE
++

MINIMUM LEVEL FLIGHT SPEED = 0.62678D+02 FT/SEC
LIFT COEFFICIENT = 0.31858D+01   DRAG COEFFICIENT = 0.41748D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.28105D+03 FT/SEC
LIFT COEFFICIENT = 0.15845D+00   DRAG COEFFICIENT = 0.19587D-0[illegible]

MAXIMUM CLIMB ANGLE = 0.56865D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.14365D+03 FT/SEC
LIFT COEFFICIENT = 0.60649D+00   DRAG COEFFICIENT = 0.33056D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.10249D+03 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.10079D+05 FT-LBS/SEC
LIFT COEFFICIENT = 0.11916D+01   DRAG COEFFICIENT = 0.74400D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.13488D+03 FT/SEC
LIFT COEFFICIENT = 0.68796D+00   DRAG COEFFICIENT = 0.37200D-01

SERVICE CEILING = 0.21460D+05 FT
VELOCITY AT SERVICE CEILING = 0.18932D+03 FT/SEC
LIFT COEFFICIENT = 0.68818D+00   DRAG COEFFICIENT = 0.37212D-01

ABSOLUTE CEILING = 0.24242D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.19123D+03 FT/SEC
LIFT COEFFICIENT = 0.74279D+00   DRAG COEFFICIENT = 0.40283D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD
0.00000D+00	0.16380D+02	0.18435D+03	0.44665D+05	0.36828D+00	0.23930D-01
0.50000D+03	0.15983D+02	0.18436D+03	0.43890D+05	0.37367D+00	0.24087D-01
0.10000D+04	0.15589D+02	0.18437D+03	0.43125D+05	0.37914D+00	0.24249D-01

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.46627D-09	0.12936D+05	0.12936D+05	0.62678D+02
0.18667D+01	0.14810D+05	0.11870D+05	0.70000D+02
0.41647D+01	0.17441D+05	0.10882D+05	0.80000D+02
0.62368D+01	0.20137D+05	0.10314D+05	0.90000D+02
0.81190D+01	0.22875D+05	0.10088D+05	0.10000D+03
0.98259D+01	0.25634D+05	0.10158D+05	0.11000D+03
0.11360D+02	0.28392D+05	0.10500D+05	0.12000D+03
0.12715D+02	0.31127D+05	0.11102D+05	0.13000D+03
0.13881D+02	0.33819D+05	0.11956D+05	0.14000D+03
0.14845D+02	0.36445D+05	0.13064D+05	0.15000D+03
0.15590D+02	0.38983D+05	0.14429D+05	0.16000D+03
0.16099D+02	0.41413D+05	0.16057D+05	0.17000D+03
0.16354D+02	0.43712D+05	0.17955D+05	0.18000D+03
0.16335D+02	0.45859D+05	0.20132D+05	0.19000D+03
0.16021D+02	0.47832D+05	0.22599D+05	0.20000D+03
0.15393D+02	0.49610D+05	0.25366D+05	0.21000D+03
0.14429D+02	0.51170D+05	0.28445D+05	0.22000D+03
0.13108D+02	0.52492D+05	0.31847D+05	0.23000D+03
0.11408D+02	0.53554D+05	0.35585D+05	0.24000D+03
0.93087D+01	0.54333D+05	0.39672D+05	0.25000D+03
0.67869D+01	0.54809D+05	0.44119D+05	0.26000D+03
0.38211D+01	0.54959D+05	0.48941D+05	0.27000D+03
0.38912D+00	0.54762D+05	0.54149D+05	0.28000D+03
0.19487D-09	0.54720D+05	0.54720D+05	0.28105D+03

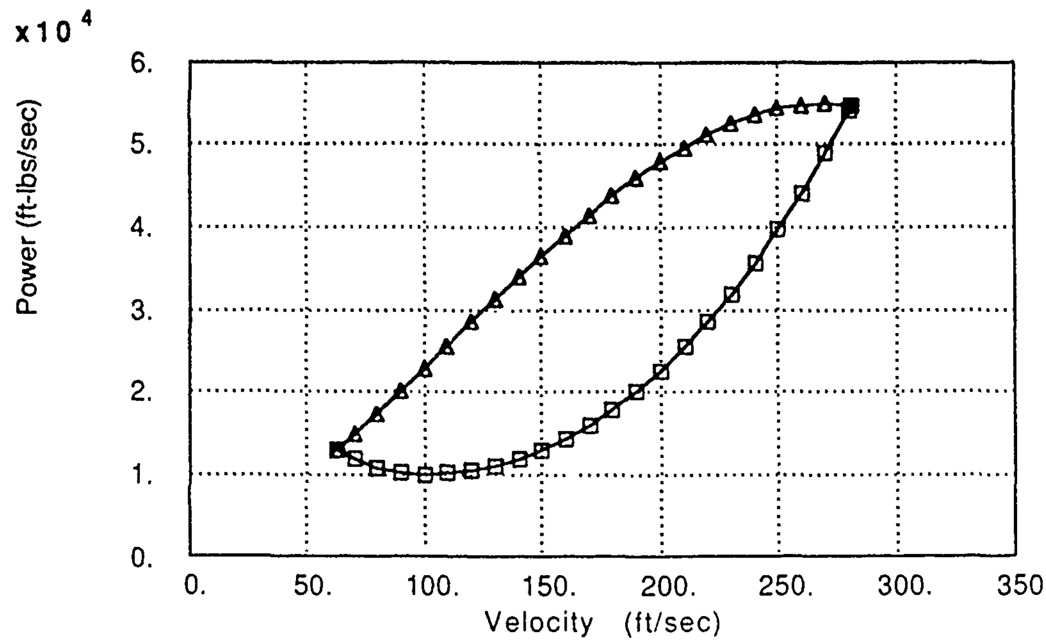


Figure H.1. Power vs Velocity Curve for the 3-Surface Configuration

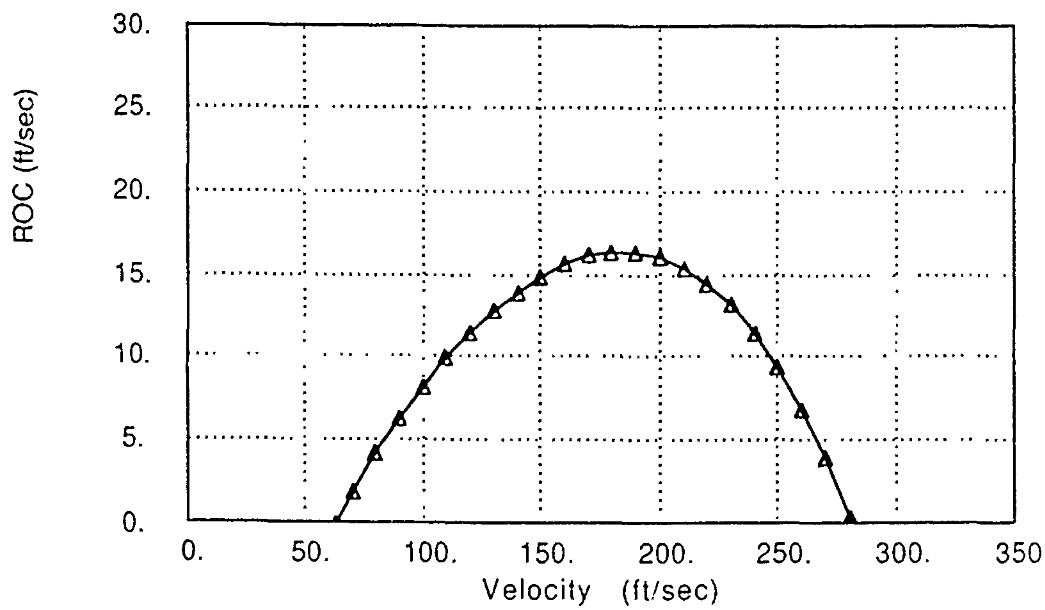


Figure H.2. Rate of Climb vs Velocity Curve for the 3-Surface Configuration

APPENDIX I MODEL FOR THE FLYING WING CONFIGURATION WITH BICYCLE GEAR

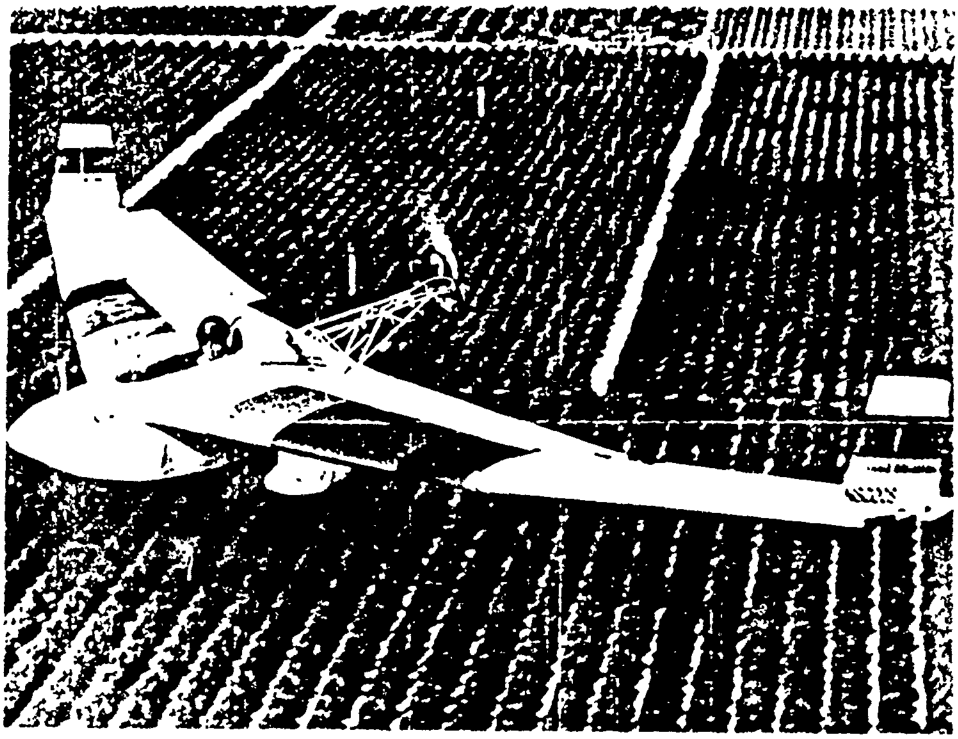


TABLE I.1. SCALED PARAMETERS BASED ON λ FOR THE FLYING WING CONFIGURATION

Flying wing aircraft based on the Mitchell U-2

Aircraft Specifications	Actual values	Scaled values $\lambda = 0.79$
Wing span (ft.)	34.00	42.84
Wing chord root		0.00
Wing chord tip		0.00
Wing area (sq. ft.)	136.00	215.89
Wing airfoil section	Wortmann mod	
Wing aspect ratio	8.50	8.50
Wing loading (GW) (lb./sq. ft.)	4.20	5.29
Vertical tail area (each)		0.00
X-section height incl canopy (ft)		0.00
X-section height (firewall)		0.00
X-section width		0.00
Length overall (ft.)	9.33	11.76
Fuel capacity (usable)	1.50	
Empty weight (lb.)	300.00	600.00
Gross weight	600.00	1200.00
Useful load	300.00	600.00
Wheel base (in.)		0.00
Nose wheel	Retractable	
Advertised performance		
Max speed sea level (mph)	85	75.73
75% cruise speed	65	57.91
Stall speed	29	25.84
ROC sl (ft/min)	500	445.45
Range at 75% (no reserve) nm	39	30.95
G limits at gross weight	10	
Horsepower	20	44.90

TABLE I.2. DRAG POLAR ESTIMATION FOR THE FLYING WING CONFIGURATION

DRAG POLAR BUILDUP

Aircraft **Flying wing RG** *Input italicized data*

Wing for Wortmann airfoil (FX 66-S-196)
Cdmin = *0.007* S = *215.9*

Fuselage (ft.)

Length = *11.8* W/L = .321
Firewall height = *3.78* H/L = .321
Max height incl canopy = *5.7*
Max width = *3.8*
Firewall X-section (sq ft) = 14.3
adjust for roundness factor of. *0.8*
Adjusted X-section (sq ft) = 11.4
% for canopy = .50
Adjusted X-section (sq ft) = 17.1

from Nicolai Figure 8.1. $cd\pi$ = *.040*

Gear from Smetana Table 5-4. $Cd\pi A\pi$ = *0.51*

Winglet rudders from Smetana Table 5-3. $cd\pi$ = *0.0058*

Component	$Cd\pi$	$A\pi$	$Cd\pi A\pi$
Wing	.0070	215.9	1.51
Fuselage	.0400	17.1	0.69
Gear			0.51
Winglet rudders	.0058	*7.6*	0.04
Total			2.75

Interference effects *0.06*
Protuberance effects *0.03*

Equiv. flat plate area (sum of $Cd\pi A\pi$ and effects) **3.00**

CDO = .01131 AR = *8.5*
e = *0.93* k1 = .0113
k3 = .0403

CD = .0113 + .0403 CL^2

TABLE I.3. POINT.TXT OUTPUT FOR THE FLYING WING CONFIGURATION

POWER AVAILABLE VS. VELOCITY
REFERENCE ALTITUDE = 0.00000D+00 FEET

PA(FT-LBS/SEC)	V(FT/SEC)
0.00000D+00	0.00000D+00
0.22875D+05	0.10000D+03
0.45859D+05	0.19000D+03
0.54762D-05	0.28000D+03
0.28944D+05	0.38000D+03

AIRCRAFT CHARACTERISTICS

CD = 0.11300D-01 + 0.00000D+00*CL**2 + 0.40300D-01*CL** 0.20000D+01
WING AREA = 0.21590D+03 SQ.FT   WEIGHT = 0.12000D+04 LBS

STATIC PERFORMANCE AT AN ALTITUDE = 0.00000D+00 FT
WITH MINIMUM TIME AND MOST ECONOMICAL CLIMB SCHEDULES TO A FINAL ALTITUD
++

MINIMUM LEVEL FLIGHT SPEED = 0.35417D+02 FT/SEC
LIFT COEFFICIENT = 0.37235D+01   DRAG COEFFICIENT = 0.57004D+00

MAXIMUM LEVEL FLIGHT SPEED = 0.26508D+03 FT/SEC
LIFT COEFFICIENT = 0.66472D-01   DRAG COEFFICIENT = 0.11478D-01

MAXIMUM CLIMB ANGLE = 0.85675D+01 DEG
VELOCITY FOR MAXIMUM CLIMB ANGLE = 0.11431D+03 FT/SEC
LIFT COEFFICIENT = 0.35742D+00   DRAG COEFFICIENT = 0.16448D-01

VELOCITY FOR MAXIMUM ENDURANCE = 0.71362D+02 FT/SEC
POWER FOR MAXIMUM ENDURANCE = 0.42203D+04 FT-LBS/SEC
LIFT COEFFICIENT = 0.91716D+00   DRAG COEFFICIENT = 0.45200D-01

VELOCITY FOR CLASSICAL MAXIMUM RANGE = 0.93918D+02 FT/SEC
LIFT COEFFICIENT = 0.52953D+00   DRAG COEFFICIENT = 0.22600D-01

SERVICE CEILING = 0.29307D+05 FT
VELOCITY AT SERVICE CEILING = 0.16158D+03 FT/SEC
LIFT COEFFICIENT = 0.46543D+00   DRAG COEFFICIENT = 0.20030D-01

ABSOLUTE CEILING = 0.32573D+05 FT
VELOCITY AT ABSOLUTE CEILING = 0.16151D+03 FT/SEC
LIFT COEFFICIENT = 0.52583D+00   DRAG COEFFICIENT = 0.22443D-01

MAXIMUM RATE OF CLIMB SCHEDULE FROM 0.00000D+00 FT TO 0.10000D+04 FT

H(FT)	R/C(FT/SEC)	V(FT/SEC)	P(FT-LBS/SEC)	CL	CD
0.00000D+00	0.21522D+02	0.16836D+03	0.41023D+05	0.16478D+00	0.12394D-01
0.50000D+03	0.21079D+02	0.16825D+03	0.40284D+05	0.16742D+00	0.12430D-01
0.10000D+04	0.20640D+02	0.16815D+03	0.39554D+05	0.17012D+00	0.12466D-01

MAXIMUM R/C, POWER AVAILABLE, & POWER REQUIRED VS VELOCITY
AT 0.00000D+00 FT

R/C(FT/SEC)	PA(FT-LBS/SEC)	PRQ(FT-LBS/SEC)	V(FT/SEC)
-0.15614D-08	0.65065D+04	0.65065D+04	0.35417D+02
0.14027D+01	0.75159D+04	0.58327D+04	0.40000D+02
0.41209D+01	0.98255D+04	0.48804D+04	0.50000D+02
0.65602D+01	0.12264D+05	0.43917D+04	0.60000D+02
0.88227D+01	0.14810D+05	0.42226D+04	0.70000D+02
0.10943D+02	0.17441D+05	0.43099D+04	0.80000D+02
0.12926D+02	0.20137D+05	0.46262D+04	0.90000D+02
0.14761D+02	0.22875D+05	0.51620D+04	0.10000D+03
0.16430D+02	0.25634D+05	0.59176D+04	0.11000D+03
0.17911D+02	0.28392D+05	0.68990D+04	0.12000D+03
0.19176D+02	0.31127D+05	0.81158D+04	0.13000D+03
0.20199D+02	0.33819D+05	0.95798D+04	0.14000D+03
0.20951D+02	0.36445D+05	0.11304D+05	0.15000D+03
0.21400D+02	0.38983D+05	0.13303D+05	0.16000D+03
0.21517D+02	0.41413D+05	0.15592D+05	0.17000D+03
0.21271D+02	0.43712D+05	0.18186D+05	0.18000D+03
0.20631D+02	0.45859D+05	0.21102D+05	0.19000D+03
0.19564D+02	0.47832D+05	0.24355D+05	0.20000D+03
0.18040D+02	0.49610D+05	0.27962D+05	0.21000D+03
0.16025D+02	0.51170D+05	0.31940D+05	0.22000D+03
0.13489D+02	0.52492D+05	0.36305D+05	0.23000D+03
0.10399D+02	0.53554D+05	0.41075D+05	0.24000D+03
0.67224D+01	0.54333D+05	0.46266D+05	0.25000D+03
0.24275D+01	0.54809D+05	0.51896D+05	0.26000D+03
0.95818D-10	0.54927D+05	0.54927D+05	0.26508D+03

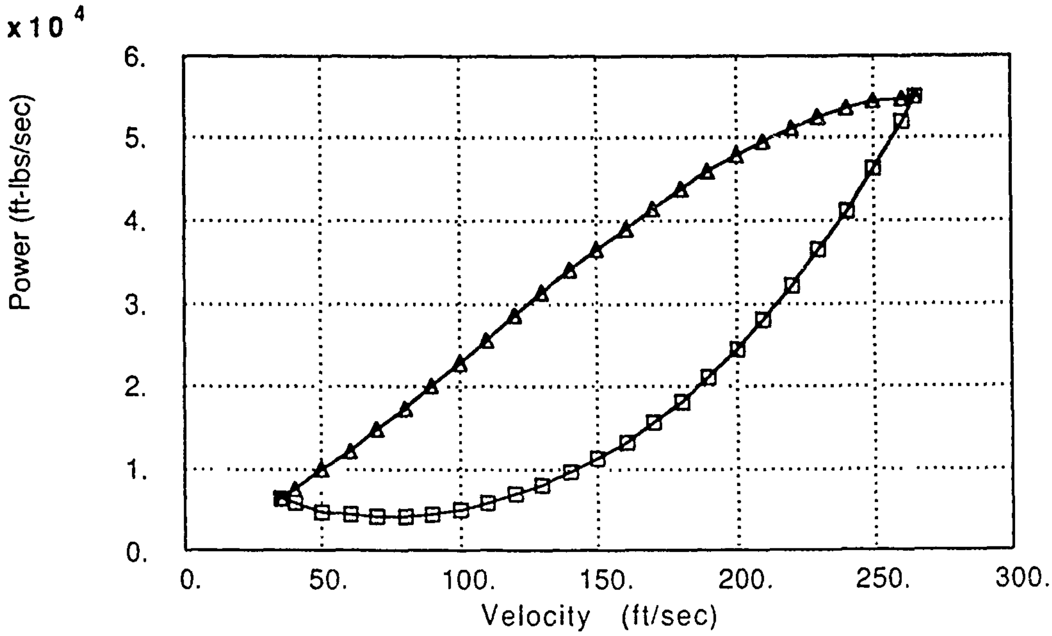


Figure I.1. Power vs Velocity Curve for the Flying Wing Configuration

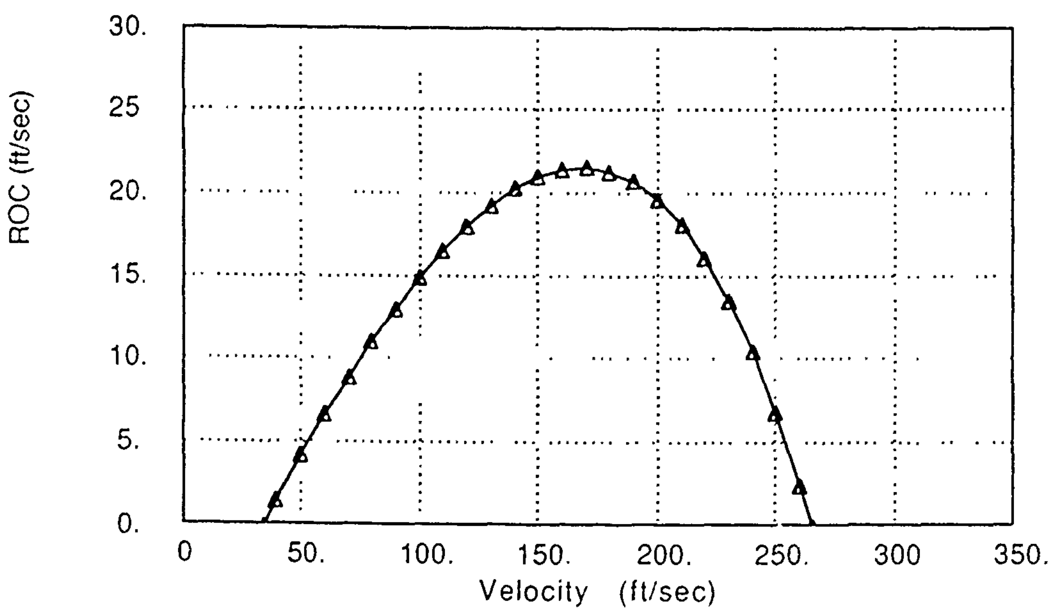


Figure I.2. Rate of Climb vs Velocity Curve for the Flying Wing

LIST OF REFERENCES

1. Lennon, A. G., *Canard, A Revolution in Flight,* AViation Publishers, Hummelstown, PA, 1984.

2. Beech Starship 1 Promotional Brochure, Beech Aircraft Corporation, Witchita, KS, 1986.

3. Wilkinson, S., "Piaggio", Air & Space, August/September 1988.

4. Heppenheimer, T. A., "The LearFan Saga", Air & Space, October/November 1988.

5. Kendall, E. R., "The Minimum Induced Drag, Longitudinal Trim and Static Longitudinal Stability of Two-Surface and Three-Surface Airplanes", AIAA-84-2164, August 1984.

6. Selberg, B. P. and Rokhsaz, K., "Aerodynamic Tradeoff Study of Conventional, Canard, and Tri-Surface Aircraft Systems", AIAA-85-4071, October, 1985.

7. "Piaggio Avanti, Beech Starship Offer Differing Performance Characteristics", Aviation Week & Space Technology, October 2, 1989.

8. Roskam, J., *Airplane Design. Part II: Preliminary Configuration Design and Integration of the Propulsion System,* Roskam Aviation and Engineering Corporation, Ottawa, KS, 1989.

9. Smetana, F. O., *Aircraft Performance, Stability and Control,* Aircraft Designs, Inc., Monterey, CA, 1988.

10. Kroo, I., *LinAir for the Macintosh,* Desktop Aeronautics, Stanford, CA, 1987.

11. *Avco Lycoming Detail Specification 249[illegible] for the O-235 engine,* Textron Lycoming, Williamsport, PA, February 1982.

12. Hartman, E. P., and Biermann, D., "The Aerodynamic Characteristics of Full-Scale Propellers Having 2, 3, and 4 Blades of Clark Y and R.A.F. Airfoil Sections", *NACA Technical Report 640*, November 9, 1937.

13. Stout, E. G., "Development of High Speed Water-Based Aircraft", Journal of Aeronautical Sciences, August 1950.

14. Hall, S., "Dynamic Modeling", Sport Aviation, July 1987.

15. Raymer, D. P., *Aircraft Design: A Conceptual Approach,* AIAA, Inc., Washington, D.C., 1989.

16. Nicolai, L. M., *Fundamentals of Aircraft Design,* METS, Inc., San Jose, CA, 1984.

17. Somers, D. M., "Design and Experimental Results for a Flapped Natural-Laminar-Flow Airfoil for General Aviation Applications", NASA Technical Paper 1865, June 1981.

18. Miley, S. J., *A Catalog of Low Reynolds Number Airfoil Data for Wind Turbine Applications,* Texas A&M University, College Station, TX, February 1982.

19. Abott, I. H., and Von Doenhoff, A.E., *Theory of Wing Sections,* Dover Publications, Inc., New York, NY, 1949.

20. Hoerner, S. F., *Fluid Dynamic Drag*, Published by the Author, 1958.

21. Smetana, F. O., *Computer Assisted Analysis of Aircraft Performance, Stability, and Control*, McGraw-Hill Book Co., New York, NY, 1984.

22. Anderson, J. D., *Introduction to Flight,* McGraw-Hill Book Co., New York, NY, 1984.

23. Cavin, D., "The Ligeti Stratos...", Sport Aviation, December, 1986.

24. Taylor, M., Taylor Mini-Imp Information Package, Longview, WA.

25. *Jane's All The World's Aircraft 1985-86,* Jane's Publishing Co. Ltd., London, England, 1986.

26. Hollmann, M., *Modern Aircraft Design,* Aircraft Designs, Inc., Monterey, CA, 1986.

27. Lyons, D., *Aerodynamic Analysis of a U.S. Navy and Marine Corps Unmanned Air Vehicle,* Master's Thesis, Naval Postgraduate School, Monterey, CA, March 1989.

28. Etkin, B., *Dynamics of Flight,* John Wiley & Sons, New York, NY, 1982.

29. Scott, W.B.,"AT3 Demonstrates Feasibility of Cargo STOL With Long Range", Aviation Week & Space Technology, September 4, 1989.

30. *Low Altitude/Airspeed Unmanned Research Aircraft (LAURA) Preliminary Development,* U.S. Naval Research Laboratory Report, November, 1986.

31. McCormick, B.W., *Aerodynamics, Aeronautics, and Flight Mechanics,* John Wiley & Sons, New York, NY, 1979.

INITIAL DISTRIBUTION LIST

		No. Copies
1.	Defense Technical Information Center Cameron Station Alexandria, VA 22304-6145	2
2.	Library, Code 0142 Naval Postgraduate School Monterey, CA 93943-5100	2
3.	Chairman, Code 67 Department of Aeronautics and Astronautics Naval Postgraduate School Monterey, CA 93943-5000	2
4.	Professor R.M. Howard, Code 67Ho Department of Aeronautics and Astronautics Naval Postgraduate School Monterey, CA 93943-5000	7
5.	Professor J.P. Hauser P.O. Box 3021 Boulder, CO 80307	1
6.	LCDR Gary D. Black Training Department PCU USS George Washington (CVN-73) 3111 West St. Newport News, VA, 23607	3
7.	Martin Hollmann Aircraft Designs, Inc. 25380 Boots Road Monterey, CA 93940	1
8.	LT Eric Pagenkopf, Code 67Pa Department of Aeronautics and Astronautics Naval Postgraduate School Monterey, CA 93943-5000	1

9. David W. Lewis 1
UAV Joint Project Office
Advanced Development Projects
Code PDA14-UDID
Washington, D.C. 20361-1014